

GRUMMAN DESIGN # 378

APOLLO EXTENSION SYSTEM

EARTH ORBIT MISSION STUDY (U)

FINAL REPORT

TO

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
ADVANCED SPACECRAFT TECHNOLOGY DIVISION
HOUSTON, TEXAS 77058

BY GRUMMAN AIRCRAFT ENGINEERING CORP.

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1 EXPERIMENT IDENTIFICATION AND DESCRIPTION

1.1 STUDY APPROACH

The primary objective of this portion of the study was to define the proposed experiments, in as complete detail as necessary, to permit their integration into Earth Orbiting LEM Lab missions.

The study guidelines were:

- Contractor to accept the initial experiment definitions provided by NASA.
- Contractor to expand the definitions adequately to permit integration studies (The initial objectives and scope definitions are to be retained. The individuals who proposed the experiment may be contacted through NASA Headquarters for additional data, as necessary.)
- Contractor to summarize the expanded and/or modified definitions as used in the integration studies, on NASA standard Experiment Identification and Description Form.

A total of eighty-six (86) experiments were selected and assigned by NASA to missions as indicated in Table 1-1. These experiments are classified into the following categories:

- 0100 - Medicine
- 0200 - Behavioral
- 0300 - Creation of Artificial Gravity
- 0400 - Bioscience
- 0500 - Space Environment
- 0600 - Liquid/Gas and Solid Behavior
- 0700 - Astronomical Observations and Techniques
- 0800 - Remote Sensing of the Earth's Atmosphere
- 0900 - Remote Sensing of the Earth's Surface
- 1000 - Maneuverable Sub-Satellite
- 1100 - Launch of an Unmanned Satellite
- 1200 - Electromagnetic Propagation and Transmission
- 1300 - Space Structures Technology

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Mission Parameters	Flight No. Altitude (n.m.) Inclination, degrees Duration, days	209 200 28.5 14	211 200 28.5 30	507 200 90 14	509 19,350 0 14	215 200 50 14	513 200 31.5 14	218 200 28.5 45	217 200 28.5 45	221 200 23.5 45	516 19,350 0 45	518 200 -83 45	521 19,350 0 45	523 200 24.5 45	236 200 28.5 45	237 200 28.5 45
Experiment No.																
0101 & 0102 - Medical & Behavioral						X		X ¹								
0103 thru 0203 - Medical & Behavioral		X	X	X	X	X	X	X		X	X	X	X	X		
0301 - Artificial G by Rotating Station						X				X						
0302 - On-Board Centrifuge									X							
0401 - Genetic Effects in Microorganisms									X		X				X	
0402 - Effects of Space on Unicell Org & Animal Tissue									X		X				X	
0403 - Effects of Weightlessness in the Primates									X						X	
0404 - Limb Regeneration & Wound Healing									X						X	
0405 - Effects of Drugs on Mammalian Behavior									X						X	
0406 - Space Borne Microorganisms									X		X				X	
0501 - Radiation Environment Monitoring		X	X	X	X	X	X	X		X	X	X	X	X		
0502 - Study of Magnetic Field Lines							X						X			
0503 - Analysis of Comet-like Particulate Clouds							X						X			
0504 - Micrometeoroid Collection													X	X		
0601 - Capillarity Studies		X						X						X		
0602A - Kinetics & Dynamics of Vapor/Gas Bubbles								X						X		
0602B - Liquid Drop Dynamics Study								X						X		
0603A - Pool Boiling in Long Term Zero G Environment								X						X		
0603B - Nucleate Condensation of Fluids in Zero G								X						X		
0604 - Density of Gradient of Fluid at Near Critical St								X						X		
0605 - Crystallization Studies								X						X		
0606 - Cosmic Ray Emission								X						X		
0701 - Emission Line Radiometry											X			X		
0702 - Intermediate size Reflecting Telescope											X			X		
0703 - Manned Coronagraph											X			X		
0704 - Nearby Solar-Like Stars in X-Rays											X			X		
0705A - Radio Astronomy 1-5 MCPS Range														X		
0705B - Radio Astronomy 1-5 MCPS using "V" Antenna					X						X			X		
0801 - Conjugate Aurora and Airglow					X						X		X			
0802A - Operation of E-R IF Radiometer Det. Equip.							X					X				
0802B - Test of IF Scanning Spectrometer							X					X				
0802C - Calibrator & Eval. of Microwave Spectrometer							X					X				
0802D - Test of Prototype Star-Tracker							X					X				
0901 - Multispectral Target Characteristics				X		X						X				
0902 - Synoptic Earth Mapping				X		X						X				
0903 - Multifrequency Radar Imagery												X				
1001 - Small Maneuverable Satellite											X					
1101 - Launch of an Unmanned Satellite (OGO)													X			
1201 - Measurement of Radio Frequency Radiation				X												
1202 - Wide Band Width Transmission in Space				X												
1301 - Effects of Space Environment on Structures														X		
1302 - Deployment of RF Reflective Structures					X										X	
1303 - Extendable Rod Performance Tests					X						X					
1304 - Ops of Solar Sailing Pass. Comp. Satellite					X						X					
1305 - Deployment & Ops. of Gravity Gradient																X
1306 - Large Aperture Space Erectable Antenna					X										X	
1401 - Personal Hygiene & Food Technology			X											X		
1402 - Cabin Atmosphere for LSS			X													X
1403 - Fluid Management Technology for LSS													X			
1404 - Radioisotope-Thermoelectric Power System Int.													X			
1405 - Orbital Test of Large Solar Cell Array				X								X				
1406 - Optical Technology													X			
1407 - On-Board Navigation & Guidance								X								
1501 - Evaluation of A4V Space Suit Assemblies		X							X							
1502 - Development of Manual Locomotion & Maneuvering		X							X							
1503 - Emergency Techniques Exp & Proc for Rescue Ops		X							X							
1504 - Personnel & Cargo Transfer Operations									X						X	
1505 - Maintenance & Repair Techniques									X							
1506 - Propellant Handling Techniques																X
1507 - Extravehicular Assembly Operations							X									
1601 - Orbital Maneuvering & Docking (Phase I & II)					X											
1602 - Observation of Echo Satellite							X									
1603 - Recapture of Syncom III					X											
1604 - Lunar Survey Camera Checkout			X													

NOTES: 1. Experiment 1.1 & 0102 Experiment Carried on Flt. 218; Experiment is Performed on Flt. 219.

Table 1-1 NASA Experiment Distribution

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- 1400 - Subsystem Development and Test
- 1500 - Extravehicular Operations
- 1600 - Maneuvering and Docking
- LOS - Qualification Test of Lunar Orbit Survey Systems

Each experiment definition was expanded by Grumman to establish a conceptual design, including:

- Performance capabilities
- Crew requirements
- Power requirements and/or profile
- Weight, size and other significant physical characteristics
- Environmental requirements
- Controls and displays
- Data handling requirements
- Where applicable, drawings and sketches showing the recommended location and construction of the experiment.

These data have been summarized on a standard experiment identification and description format and are included as Volume 4 of this report.

In addition to the experiments defined by NASA, a number of Air Force MOL experiments were integrated into the LEM Lab. In most cases the Air Force MOL experiment descriptions were utilized as defined. Volume 3, Appendix A, presents the supplementary data used to integrate the experiments into the LEM Lab. These data include equipment parameters such as: volume, weight and power requirements; display layouts; and sketches of equipments.

1.2 BIOMEDICAL/BEHAVIORAL EXPERIMENT PROGRAM

The first 30-day and 45-day missions are considered primary biomedical/behavioral flights. On these flights (211 and 218), a complete Biomedical/Behavioral Program has been scheduled with priority over other engineering/scientific experiments. However, this program differs from that presented in Volume 4. That program requires a total man hour schedule approximately 50 percent in excess of the available time. The available time being based on a work schedule of ten hours per man and a three-man crew.

The reduction in scheduled man hours required for the Biomedical/Behavioral Program has been achieved with no significant degradation to the experiment program. This has been accomplished by judicious combination, substitution, and/or elimination of experiments based on the following rationale:

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- Commonality of equipment and procedures for testing the same and related physiological systems.
- Substitution of related experiments, alternating experiments in the prescribed frequencies during the extended missions of 30 and 45 days.
- Elimination of hazards to the health of the astronaut resulting from using the specified experimental procedure. Pertinent data points for these experiments are "fall outs" of other specified experiments.

The experiments involved in this modified program, and the reasoning used in the modification, are listed below:

Experiment	Rationale
0101 and 0102	<u>Artificial "g" experiments</u> • Will be performed on Flights 219 and 221 only
0109 and 0110	<u>Respiration experiments</u> • Commonality of equipments • Integration of procedures and data collection for both can be achieved at one time.
0115 and 0116	<u>Temperature experiments</u> Alternate experiments within frequency cycle. Some 0115 data can be obtained from 0116 experiment.
0118 and 0119 and 0120	<u>Blood Experiments</u> • Blood sampling and storage common to all • Integration of microscopic procedures and data collection for both can be achieved at one time.

0111 and 0112 and 0117

Muscle, Hormones and Mineral
Metabolism experiments

- Urine collection and storage common to all
- The Electromyographic experiment can be easily integrated into the procedures.
- Blood sampling and collection of exp. 0117 can be incorporated into the 0118 experimental series above.

0114

Gastro Intestinal experiment

- Delete experiment as stated in description form.
- Information concerning integrity of the "G.I." tract can be determined from other experiment, i.e., 0113
- It has been suggested that "the string attachment will produce gross gastro-intestinal damage." (N.A.A.)

0103 and 0108

Cardio Vascular System experiment

- Commonality of Equipment
- Integration of procedures and data collection for both can be achieved at one time.

0106 and 0107

Cardio Vascular System experiment

- Commonality of equipment
- 0106 procedures can be integrated into 0107 without increasing time
- No physiological interference or masking of results

0121

Microbiological experiment

- 30 days needed for bacterial stabilization
- Recommended for 45 day and missions only

0200 series

- No changes

2 MISSION ANALYSIS

2.1 STUDY APPROACH

The overall mission analysis objective was to develop Preliminary estimates of the duty cycles for the crew and equipment on each of the twenty-one flights. For reasons of time and study commonality, the analysis was constrained to use the NASA-defined grouping of experiments by flights as exactly as possible. In all but a few cases, this was achieved.

For each flight, the experiment complement was reviewed to:

- Supply needed information to complete the experiment definition
- Determine compatibility of the various experiments, i.e., whether mutually exclusive requirements occurred; whether some portions of experiments could be combined; or whether some equipment could be shared with other experiments or with the spacecraft subsystems.

When the experiments were fully defined, their compatibility with the flight parameters (altitude, inclination, booster, etc.) was assessed. At this point the duty cycle charts were constructed, considering crew and subsystem constraints. Guidelines used in constructing these charts were as follows:

- The first flight of increased mission duration (Flights 211 and 218,) should be primarily a biomedical/behavioral flights.
- Experiments may be partially or totally combined where practical.
- At least one crewman should be awake at all times.
- All crewmen should be awake when any are extravehicular, and during propulsive maneuvers.
- At least one crewman should be specifically assigned to the Lab as standby monitor during EVA's.
- Each crewman should be scheduled as closely as possible for four hours on duty followed by four hours off duty, with a maximum of ten hours on duty per twenty-four.
- The minimum continuous sleep period per man should be four hours, with an objective of six to eight uninterrupted hours for eating, personal hygiene and sleeping.
- Maximum extravehicular times are: 3.5 hr on PLSS; 1 hr on standard AMU
- Arrangement of off duty times should, where possible, provide concurrent rest periods between two crewmen, rotating these shared times among all crewmen.

- Extravehicular experiments should be scheduled to coincide with the seven-day requirement to replenish lithium hydroxide whenever possible.
- Both PLSS units must be fully charged before EVA.
- During EVA, the crewman in the Lab is suited and on the ECS "suit loop" when the cabin is depressurized.

The results are presented in Tables 3-1, 3-2 and 3-3 for each flight in Volumes 2 and 3. The experiment mission profile, Table 3-3, indicates power, stability and pointing accuracy requirements versus time for the experiment equipment only. Either a complete profile or selected days are presented; whichever is more appropriate to the particular mission. The crew duty charts, Table 3-2, show all non-identical days.

Generally, the crew experiment equipment time charts begin with the first experiment period. The times to achieve orbit and return to earth are accounted for by leaving a portion of the last day of the mission as unassigned time. This technique simplifies the task of crew scheduling and is believed reasonable for this study considering that the emphasis here is on orbital operations. Where orbit achievement and experiment times overlap, e.g., Flight 521, the time chart begins at launch.

In keeping with this study's emphasis on experiments, their integration and operation, no specific times are indicated for spacecraft systems checks. "Housekeeping" activities such as status monitoring, fuel cell purging, and IMU updating, occupy a small time per day and adequate unassigned time is available. In-orbit repairs, while potentially requiring significant time, cannot be anticipated and scheduled, but again, the unassigned time allows some flexibility to accommodate them.

One other simplifying assumption should be noted. For the low-altitude flights, a 90-minute orbital period was used. Since the actual period is slightly longer, there will be approximately 15.5 orbits per day instead of the sixteen shown. It is assumed that where earth dark/light or spacecraft occulted/sunlit situations are significant to the experiment (indicated by marginal symbol on Table 3-2), the schedule would be adjusted if necessary to coincide with the nearest occurrence of the desired position in the orbit.

The mission analysis for each flight is summarized in Volumes 2 and 3 in Table 3-1. It serves as a guide to the crew schedules, experiment by experiment, shows the total man-hours expended on experiments, the distribution of work load among the crew members, the degree to which experiment time objectives are achieved, the extremes of work day fluctuation, and other pertinent mission information. "Specified" experiment times in the table refer to the times presented in Volume 4 of this report.

The results of the mission analysis for the experiments were combined with other spacecraft requirements to accomplish the subsystem designs. In conducting this study, the areas of mission analysis, subsystems design and configuration definition (spacecraft design and integration) do not form a simple sequence, but an iterative loop, with an additional feedback from configuration into experiment-spacecraft compatibility. These relationships are shown in Figure 2-1.

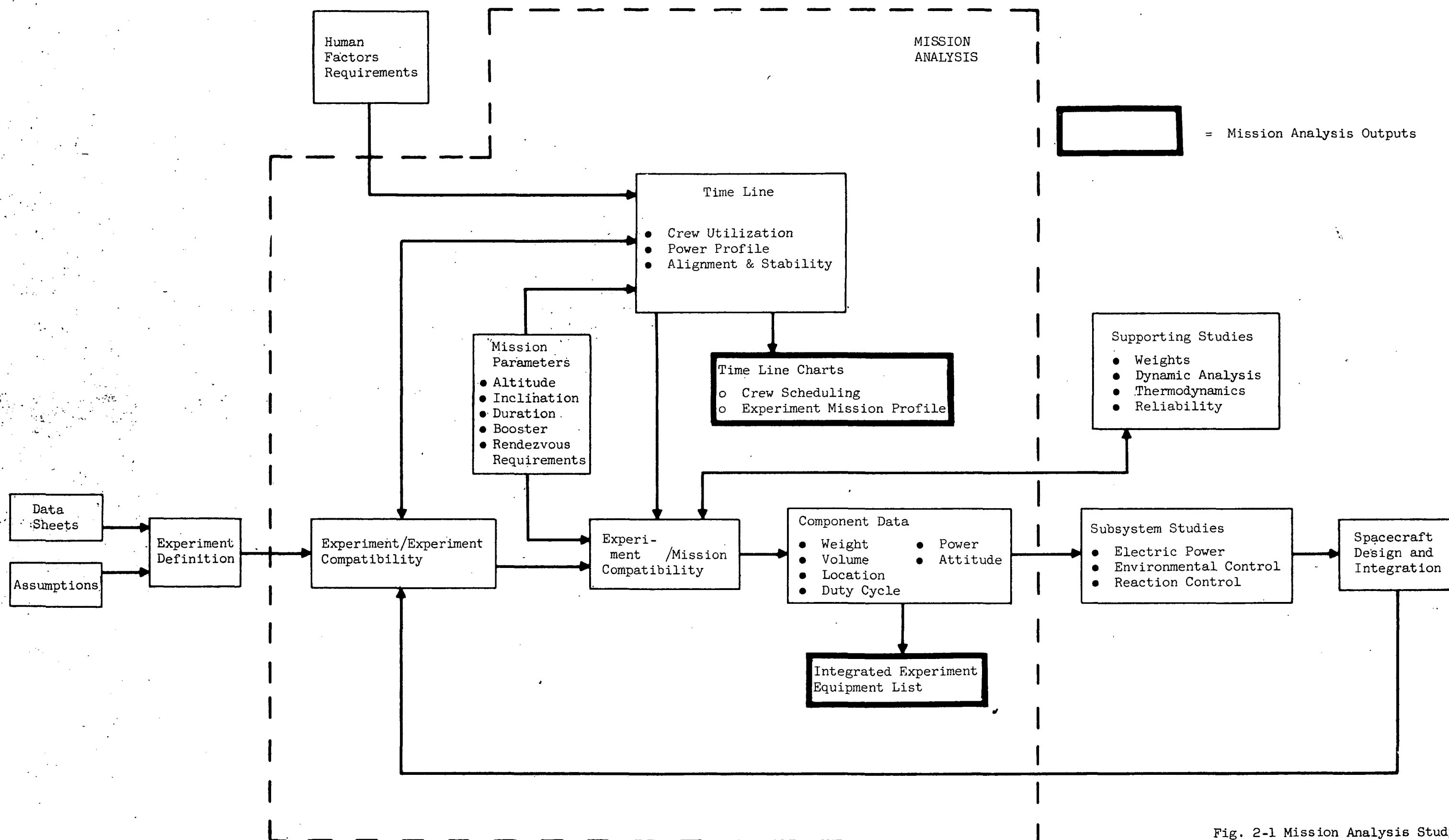


Fig. 2-1 Mission Analysis Study Approach

3 SUBSYSTEM STUDIES

3.1 POWER SUPPLY AND DISTRIBUTION SYSTEM

3.1.1 General

The power supply and distribution system consists of the fuel cell assemblies and reactant tanks, peaking battery, inverters, and the power distribution equipment. The reactant oxygen is housed in the same tankage as the life support oxygen.

For the "C" configuration the power supply is sized to provide only "LEM Housekeeping" and experiment power for the entire mission duration. For the "D" configuration, however, the power supply and tankage has been made sufficient to accommodate these requirements plus CSM Housekeeping and spacecraft life support beyond that which is provided by a standard Block II SM. The time at which the Lab "takes over" this additional requirement depends upon the type of orbit flown. This is due to the differences in the times required in ascent and descent for the low inclination Earth orbit, polar orbit, and synchronous Earth orbits. Sufficient fuel cell reactants have been reserved in the service module to meet energy requirements from deorbit to entry on fuel cell power if this mode of operation is selected in lieu of batteries (special H₂ tank required for synchronous orbit cases).

The calculated Lab "take over" times are the following:

	Oxygen Takeover (Days in orbit)	Hydrogen Takeover (Days in orbit)
Low Incl Orbit	14.8	17.5
Polar Orbit	14.3	19
Synchronous Orbit	13.25	17.5

The lower takeover point (oxygen) governs for both fluids since no interchange of fuel cell reactants between lab and SM has been assumed.

Typical LEM Lab power supply average loads are shown below. A .925 DC distribution efficiency and a .72 AC conversion and distribution efficiency are included in all cases.

LEM Lab Housekeeping

E.C.S.	121
Instrumentation	32
Displays	75
E.P.S. Controls	50
Crew Systems	22
Distribution Loss	25
	325 watts

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Parasitic Power for each operating FCA	190 watts
Each single reactant tank	11 watts
CSM Housekeeping (FCA parasitic Power deleted)	1100 *watts

* Ref: Subsystem Definition for Addendum I to NAS 9-3140 updated by telephone conversation with NAA

Mammalian Life Support System (added to Flights 219 and 229 housekeeping requirements)

Fans	62.5
Suit Fan	60.0
O ₂ Sensor	1.0
Water Management	1.0
Distribution Loss	10.5
	135.0 watts

The total electrical energy supplied by the LEM Lab for each mission is shown in table 3.1-1.

3.1.2 Fuel Cell Assemblies and Reactant Tanks

For the purpose of this study the P & W 1000 hr. fuel cell was used as the basic power plant. With the exception of operating life all characteristics of the 1000 hrs. fuel cell were taken to be the same as those of the present Apollo CSM fuel cell. In-flight start up capability is assumed.

Apollo CSM Fuel Cell Characteristics

Max. Gross output Power at 31 volts or less	1420 watts
Min. Gross output Power at 27 volts or more	563 watts
Weight including support cone	247 pounds
Max. Module diameter	22.5 inches
Max. Module height	44 inches

Reactant consumption rates were taken to be those given in "Subsystems Definition for Addendum I to NAS 9-3140". For configurations I and C, power levels averaged over the mission were used to calculate reactant requirements and to determine the number of FCA's required. For missions in the D configuration, average power levels were determined for the periods before and after the takeover point and the number of FCA's and quantity of reactants required were determined on this basis. The number of FCA's required for each flight is shown in table 3.1-1. On the D configuration one additional FCA is started at the takeover point to provide CSM

housekeeping power. This FCA is stored cold to avoid the penalties in reliability and the continuous heater power associated with hot standby. A number of hours before the takeover point the additional FCA will be heated and maintained hot electrically using power from the CSM FCA's.

Fuel cell redundancy for each flight has been established from existing failure rate data (see section 4.2). These studies have shown that one spare FCA must be carried on all flights with the exception of DOD 7B, configuration D, which requires two spares. In the event of FCA failure, heatup of the spare fuel cell, which requires 650 watts for 4 hours; will be accomplished as follows:

- Configurations I & C (and configuration D before takeover).
If one FCA is operating and one is in cold standby, failure of the operating FCA will require that power from the CSM be used for start-up of the spare. A minimum of two FCA's will be operating in the CSM giving a power capability in excess of that required by CSM and LEM housekeeping (1389 watts and 325 watts) plus the 650 watts needed for the spare FCA heaters. If two or more FCA's are operating at the time of a failure, start-up of a spare can be accomplished as described above or the LAB experiment loads may be dropped from the operating FCA and it can then provide heater power to the spare.
- Configuration D after takeover.
If only two fuel cells are operating a total failure of one will require that a large part of the CSM housekeeping loads be dropped so that power is available for heating the spare. Since heater power need not be supplied continuously housekeeping equipment can be operated for short periods while the fuel cell is being brought up to operating temperature.

If three or more fuel cells are operating when a failure occurs, heater power can be supplied to a spare fuel cell while maintaining full CSM and LEM housekeeping loads.

The reactant tanks are discussed in section 4.1.2.

3.1.3 Peaking Battery

Peak power requirements in excess of the 1420 watts from each FCA are supplied by silver zinc or silver cadmium secondary batteries which are then recharged by the fuel cells during periods of lesser loads. The magnitude of peak power demands, their duration and frequency are taken from the power profile. The difference between the peak power load and the maximum fuel cell output power is then used to determine battery current and ampere hour requirements at 28 volts. Actual cell ampere hour capacity is calculated assuming a 75% depth of discharge, a maximum discharge at a ten minute rate and a maximum charge at a rate of C/10.

Flight 229 will be used as an example of battery sizing. On the first day of the flight there is a single peak experiment power requirement of 850 watts. The total peak load (including housekeeping, FCA parasitic power, reactant tank power and experiment power) is 1637 watts lasting for 2 hours. This is 217 watts more than is available from the FCA. At 28 volts 7.8 amperes and 15.6 ampere hours are required. With a 75% depth of discharge a 21 ampere hour cell is therefore needed. The only other peaks in the flight are five minute spikes ranging up to 1140 watts on days three and four. These spikes will require up to 19 amperes which can

readily be supplied by the 20 ampere hour battery. The power profile shows that there is sufficient time between spikes to recharge the battery. Only the peak of the first day represents a deep battery discharge (each of the five minute spikes represent a discharge of less than 10%) and a silver zinc battery was therefore chosen. The weight of a sealed 20 ampere hour 19 cell battery is approximately 20 pounds.

3.1.4 Inverters

The standard LEM Inverter is a single phase 350 VA unit. Because three phase power is required by the CSM, FCA hydrogen and glycol pumps (which make up most of AC load), the LEM inverters are replaced with CSM three phase 1250 VA inverters. These inverters are capable of an output many times the required power but are chosen due to their availability in the Apollo program.

Reliability requirements indicate that one spare inverter must be carried on 14 day flights and that two spares must be carried on flights of 30 and 45 days. In addition the moderate load levels relative to the full load capability of any one inverter increase the probability of successful operation.

3.1.5 The Power Distribution System

The system shown in Figure 3.1-1 consists of primary Fuel Cell Assemblies, (FCA's), 30 invertors, a battery storage system, and associated busses. All operating FCA's are connected in parallel to the LEM Lab and CSM (in the D configuration) busses while the spare FCA's remain deactivated and isolated from the power distribution system. Startup and test capability for the standby FCA's will be provided so that no power disturbance occurs at the buss.

Power from the FCA's will be transmitted by redundant multiple feeders to the LEM Lab and CSM distribution points. Indicating fuses will be at both ends of each feeder to provide isolation at the bus and at the FCA should a line to ground fault occur along a feeder. Each feeder is 3/2's over-designed so that with one feeder out and two feeders remaining, full load capability is maintained.

The battery peaking bus supplements fuel cell capacity during peak power load demands. A battery charge controller is provided to maintain battery charge.

Table 3.1-1 Electrical Energy Supplied by LEM Lab.

Flt No.	No. Of Fuel Cells		Avg Exp Pwr Delivered, Watts	kw-hr For Exp	1 & D Config kw-hr For LEM Power Pack Design	C-Config For LEM Power Pack Design
	C	D				
209	--		108	36	146	---
211	2	3	696	502	1400	907
507	2		650	218	415	---
509	2		225	76	264	---
215	3		1140	383	652	---
513	2		753	253	440	---
218	2	3	62	67	1651	680
219	2	3	162	175	1903	950
221	2	3	90	97	1799	848
516	2	3	535	578	2205	1180
518	2	3	487	525	2117	1130
521	2	3	108	117	1732	721
523	2	3	368	397	1978	1005
229	2	3	412	445	2172	1200
230	2	3	693	746	2330	1360
DOD 1	2	3	264	190	1073	594
2	3	4	1240	890	1940	1450
3	2	3	250	180	1069	586
4	2		108	36.4	232	---
5	2		173	58	247	---
6	3		835	280	550	---

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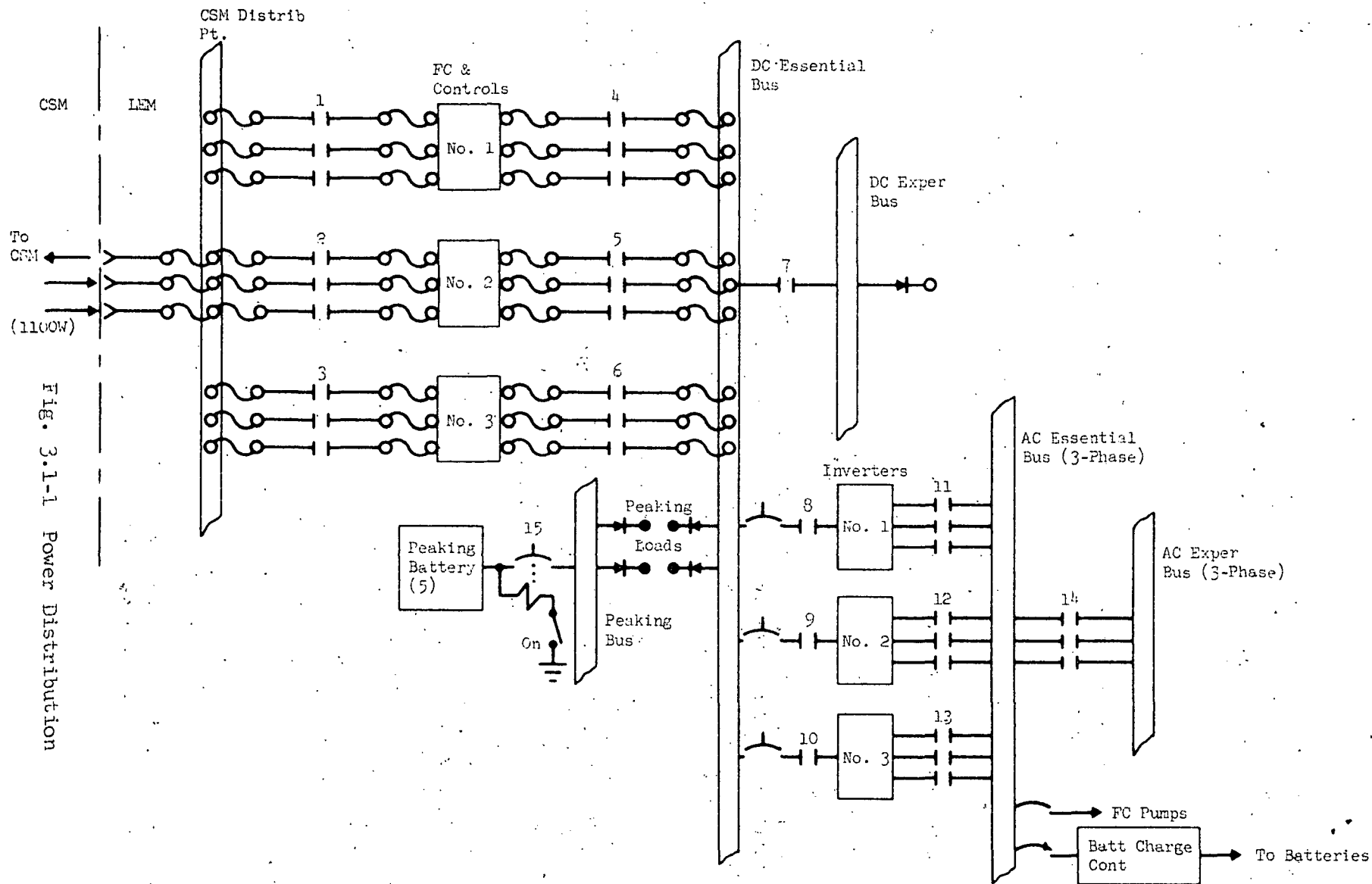


Fig. 3.1-1 Power Distribution

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3.2 LEM LAB ENVIRONMENTAL CONTROL SUBSYSTEM

Although experimental programs associated with each LEM Lab flight will impose some unique requirements on portions of the Environmental Control Subsystem, the basic configuration described below and illustrated in Figure 3.2-1 is representative of all flights, with exceptions in some areas of Flights 209, 219 and 229 which are discussed separately.

3.2.1 Basic LEM Lab ECS

3.2.1.1 Atmosphere Revitalization Section - the current LEM Atmosphere Revitalization Section (Suit Circuit) is retained on all proposed flights. It is intended that this equipment perform the following functions:

- Crew support in an unpressurized LEM Lab module during extravehicular activities and emergency umbilical support exterior to the LEM Lab cabin.
- Primary crew support on those flights where LEM Lab-CSM separation is required by the experimental or flight program.
- Environmental control functions for test animals on biological flights (see Figure 3.2-1 inserts A and B.)
- A possible means by which CSM ECS spare parts and maintenance procedures may be minimized is by sharing operating time with LEM Lab equipment. The LEM Atmosphere Revitalization Section, designed to support two suited crewmen, is capable without major modifications of supporting three men in shirtsleeves.

During routine flight, the CSM atmosphere revitalization equipment will condition the combined CSM-LEM Lab atmosphere. To insure adequate recirculation between cabins, a flexible duct is installed running from the CM, through the docking tunnel and hatches, to the LEM Cabin Recirculation Assembly, which is equipped with fans whose operating characteristics are consistent with this configuration. Atmosphere return is through the open docking tunnel and hatches.

3.2.1.2 ECS Oxygen Supply and Cabin Pressure Control Section - The LEM descent stage gaseous oxygen accumulator, sized for repressurizing the LEM cabin, is retained as are associated PLSS refill and cabin repressurization controls. Provisions for delivering oxygen to the Atmosphere Revitalization Section and maintaining cabin pressure are also retained. Integration of CSM water management equipment (discussed below) provides the capability of delivering 100 psia oxygen to the CSM for separation of metabolic water from the combined spacecraft atmosphere. This also provides a capability to continuously maintain CSM atmospheric pressure from the LEM during closed hatch operation. The oxygen docking umbilical is shown in Figure 4.7-1. LEM ascent stage GOX tanks are deleted from all flights due to their limited capacity. The descent stage GOX tank is kept in a charged condition by SOX tanks in the fuel cell reactant storage subsystem.

3.2.1.3 Water Management Section - The CSM water control module, water tanks and associated equipment are used in place of the LEM Water Management Section for the following reasons:

- CSM water management equipment, unlike that of the current LEM, is designed to collect water continuously from CSM fuel cell assemblies and condition it for crew use or disposal by overboard venting, as required during the mission. It is, therefore, considered as necessary interface equipment between CSM fuel cell assemblies (in the Lab) and the Lab ECS.
- LEM water tanks are prepressurized and undergo only one expulsion cycle during the LEM mission. They are, therefore, not designed for the cyclic demands of operation in conjunction with fuel cells.

Incorporation of oxygen pressure regulators associated with the CSM water control module provides a source of oxygen at 100 psia which the LEM Lab is required to deliver to the CSM ECS for operation of metabolic water separation equipment after CSM oxygen tanks are depleted.

Since LEM water tank pressure varies from approximately 50 psia, at fill, to near cabin pressure when depleted, CSM water tanks operated at a constant pressure of 20 psia are inherently compatible with LEM ECS water separation equipment.

3.2.1.4 Thermal Control Section - The primary mode of heat rejection will be radiators sized for the average load with a supplementary water sublimator to accommodate peaks. The water sublimator will utilize only that water which is produced by fuel cells and not required for other purposes (i.e., drinking, food preparation, personal hygiene, etc.).

The Thermal Control Section of the LEM ECS has been modified as follows for these flights:

- With the exception of Flight 209, where water sublimators are retained for heat rejection, radiator assemblies have been integrated into the Thermal Control Section. A modular radiator panel approach is planned, since area requirements vary with the experimental equipment on each flight. Panels will be located on the descent stage and integrated into the heat transport loop upstream of the primary water sublimator.
- Since the secondary heat transport loop services only that equipment which is critical in a LEM mission abort mode, and is charged subsequent to primary loop failure by the water management section, it is not considered applicable to LEM Lab missions and has, therefore, been deleted. The secondary coolant pump has been retained, since it is an integral part of the primary coolant pump assembly. Diverter valves permit this unit to be engaged manually, should the primary coolant pumps fail.
- The most variable modification of the Thermal Control Section is the addition and deletion of cold plates as dictated by experimental and support equipments required on each flight. Provisions for equipment requiring cold plates are shown in both the pressurized and unpressurized ascent stage, as well as in the descent stage.

3.2.2 LEM Lab ECS - Flight 209

Due to a relatively low total energy requirement and a more than adequate Lab weight allowance, batteries have been retained as the primary source of LEM Lab power on Flight 209. The existing Water Management Section of the LEM ECS has thus been

retained in its entirety. The storage capacity of LEM water tanks (398 lbs) is sufficient to satisfy both the requirements of the experimental program and of the Thermal Control Section which utilizes LEM water sublimators exclusively to reject LEM Lab heat. One Block II CSM cryogenic oxygen tank is incorporated to provide oxygen to repressurize the Lab following extravehicular excursions and to recharge portable life support systems. Batteries, twelve of which are located in the descent stage, and two in the ascent stage, are cooled actively by the primary coolant loop as shown in Figure 3.2-2. Cold plates for the electronics associated with the LEM Lab Flight Control Subsystem are also shown in Figure 3.2-2. These cold plates would be eliminated if 1500 Series experiments are not expanded to include astronaut recovery by a maneuverable spacecraft (see Flight 209 discussion).

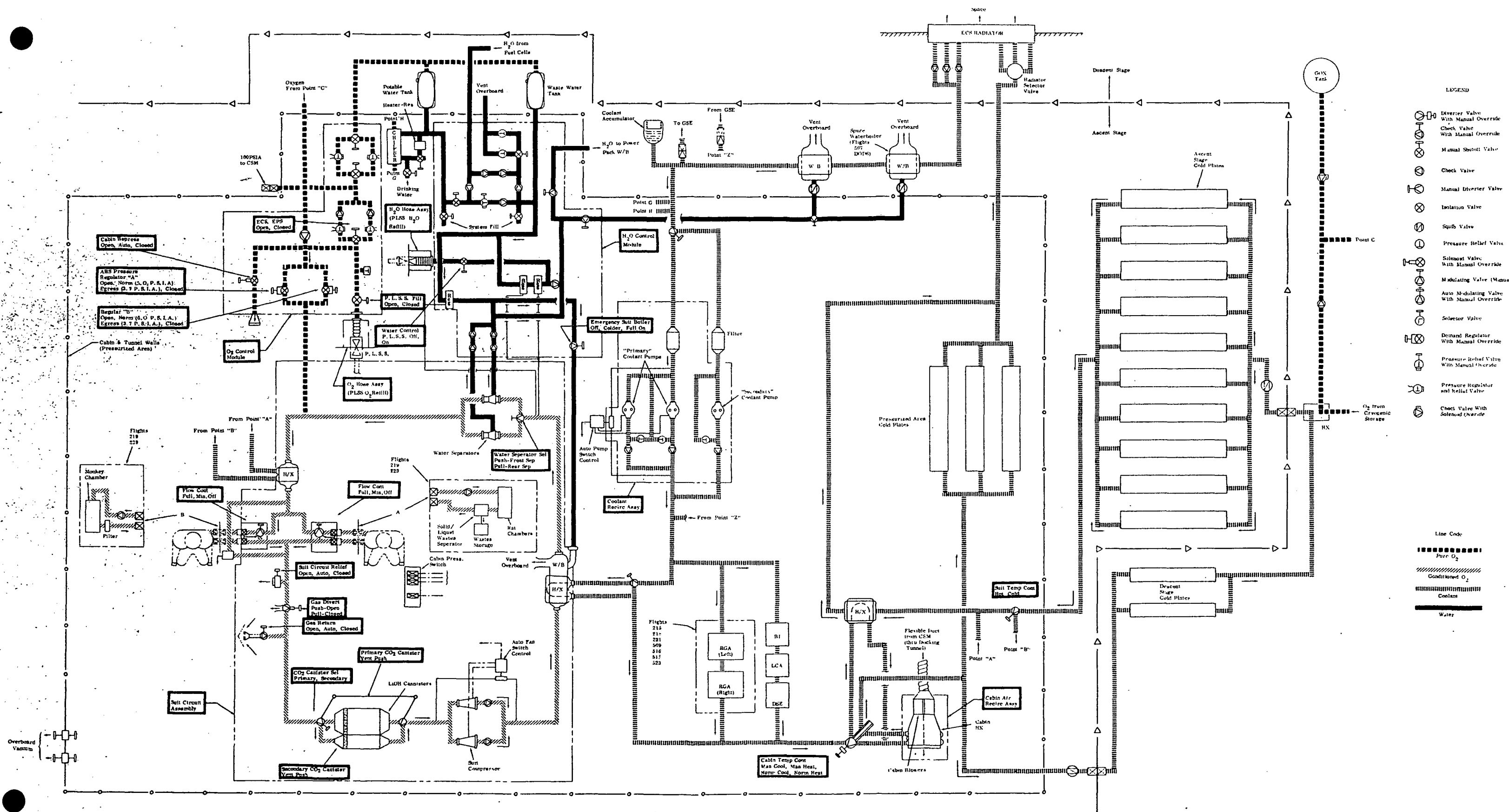
3.2.3 Environmental Control for Biological Specimens (Flights 219 and 229)

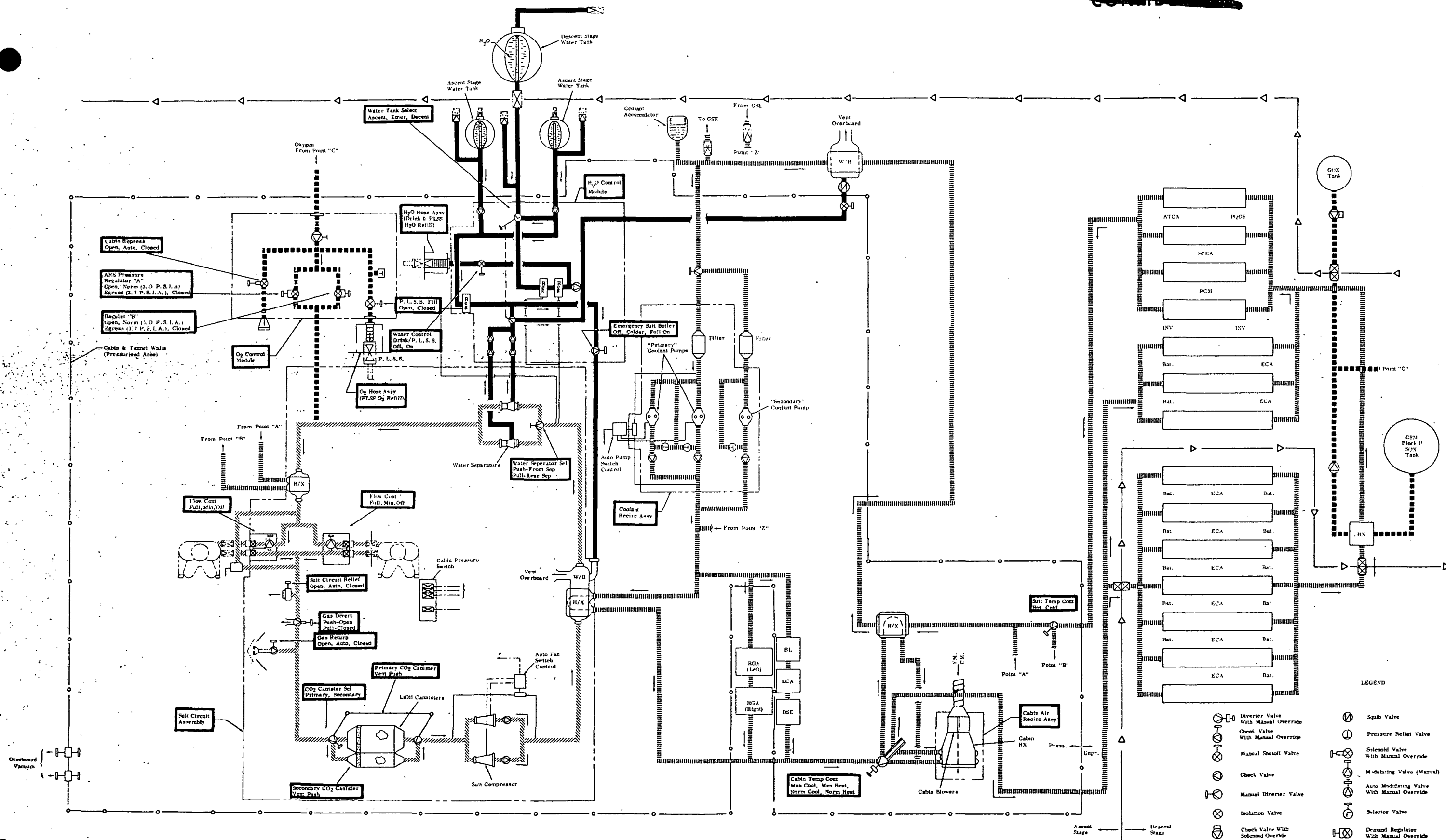
Flights 219 and 229 carry a mammalian experiment in which a chimpanzee and 50 test rats are housed in separate pressurized containers. Atmosphere revitalization for these containers is provided by the LEM suit circuit through the umbilical connections normally used by the astronauts. This is shown schematically in Figure 3.2-1. To prevent the metabolic waste products of the test rats from clogging and contaminating the suit circuit, a liquid and solid waste separator and storage unit must be provided downstream of the rat containers. Both of these assemblies are development items. It is anticipated that the transfer of waste products from individual specimen containers to separation equipment will be accomplished by maintaining flow of sufficient velocity to move the largest particles expected. Rats must also be restrained so that container inlet oxygen will be respired prior to its becoming contaminated by products of metabolism. Chimpanzee wastes are collected by means of a catheter and a fecal bag, independent of the atmosphere revitalization section. A solids filter is, however, provided downstream of the container to prevent debris from entering the airstream.

3.2.4 Fuel Cell Cooling System

Integration of CSM fuel cell assemblies into the Thermal Control Section of the LEM Lab ECS requires a radiator, a water sublimator and associated plumbing, as shown in Figure 3.2-3. The radiator serves as the normal heat sink for the fuel cell assembly's secondary glycol loop. However, during the interval in which the radiator is covered by the shroud, (lift-off until jettisoning of the shroud) the water sublimator serves as the primary heat sink. For the synchronous orbit flights, this interval is approximately of 5 hours duration. During normal flight conditions, the water sublimator will also reject peak fuel cell heat loads.

Reliability considerations dictate the need for the radiators to have three sets of redundant tubes. A selector switch is provided that will automatically switch sets of tubes in the event of a radiator puncture.

~~CONFIDENTIAL~~Fig. 3.2-1 Basic LEM Lab
ECS Schematic~~CONFIDENTIAL~~

~~CONFIDENTIAL~~Fig. 3.2-2 ECS Schematic
Flight 209~~CONFIDENTIAL~~

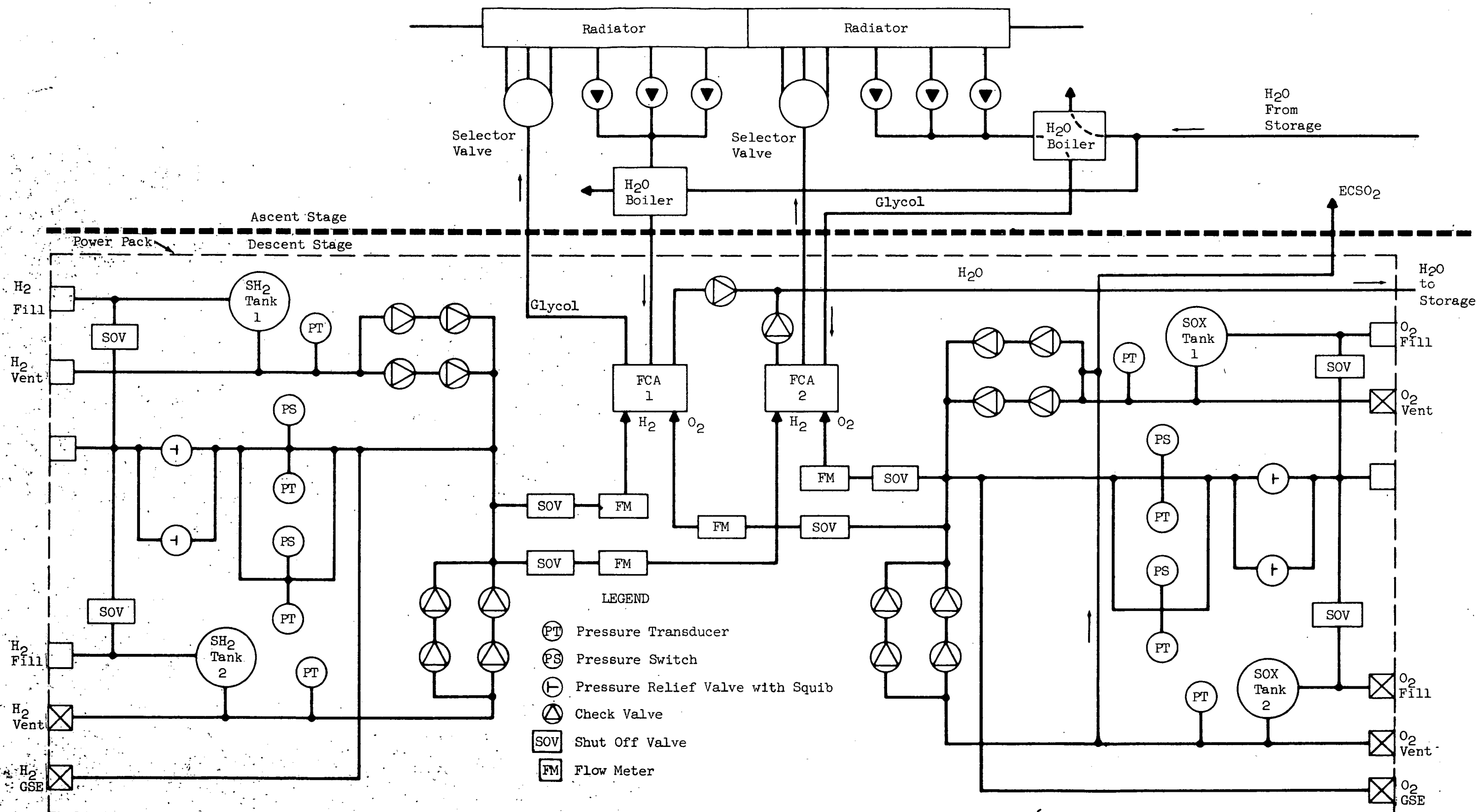


Figure 3.2-3. LEM Lab Power Pack Schematic

3.3. STABILIZATION AND CONTROL SUBSYSTEM

All flights have been examined to identify stabilization and control (SCS) requirements. These requirements have resulted in eight standardized configurations, with small variations, which make maximum use of LEM components. Flights which require Lab attitude or translation control while separated from the CSM contain a complete basic flight control system. This same system is required when Lab RCS reactants are needed to supplement SM reactants for extended flights. Additions to this basic system are provided for such functions as operation of the RCS in one Lab by the SCS in another, extended earth and inertial pointing, remote deadband control, and operation of the CSM control system from the Lab.

3.3.1 Configuration I

This basic system uses only LEM SCS components. It is designed to allow a crew member to effect attitude and translation control of the LEM Lab by means of the Lab Reaction Control System (RCS). The system will operate in a rate command mode and, in addition to Lab RCS, will include Stabilization and Control (SCS) electronics, hand controllers, and panel displays. This is indicated on the block diagram shown in Figure 3.3.-1.

Movement of the attitude controller from its neutral position will direct signals through the Attitude and Translation Control Assembly (ATCA) into the primary solenoids of the reaction jet thrust chamber assemblies (TCA). These signals command vehicle rotations about the roll, pitch and/or yaw axis. This motion will be sensed by rate gyro assemblies to provide feedback signals to the ATCA summing amplifiers, and establish vehicle rotational rates directly proportional to attitude controller displacements. Provision is also available to command pulsed or steady-on operation of the reaction jets in attitude control by appropriate selection through the SCS panel. This capability is selectable for each axis. In the event of a failure within the SCS and/or RCS, emergency switches are provided at the extremes of controller displacement to direct signals to the secondary solenoids of the reaction jet thrust chamber assemblies. Error needles will be provided on the control panel for displaying vehicle body rates.

Movement of the translation controller from its neutral position will direct control signals through the ATCA into the primary solenoids of the reaction jet TCA's in order to command open-loop linear acceleration control along the X, Y and/or Z vehicle axes. Partial movement of the translation controller will operate the reaction jets in a pulsed manner, while full displacement of the controllers will command the jets steady-on.

The SCS panel, in addition to allowing selection of pulsed, direct, or normal attitude control operation, will also provide means for selecting a minimum and maximum range of limit cycling and the disengaging of flight control operation. The minimum deadband has been taken at $\pm .2^\circ$ and the maximum has been assumed to be $\pm 5.0^\circ$. An RCS panel will be provided for monitoring propellant consumption, tank pressure, temperature, and for isolating failed jets.

This configuration is used on Flight 209 to provide attitude and translation control from the Lab during emergency rescue experiments. This configuration is also required on Flight 221, during which the Lab and CSM are separated by cables to perform artificial "g" experiments, for spin up and for vehicle control.

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Flight 218C also utilizes this configuration, except that an Experiment Sensor Package is used as an attitude reference unit (similar to Configuration III).

3.3.2 Configuration II

This control system is exactly the same as configuration I, and will be used to separate Flight 218 Lab from the CSM and to dock this Lab to Flight 219 Lab, and to control the 219 RCS after docking. This will allow use of RCS reactants in the second LEM Lab. (219-D configuration only)

The block diagram shown in Figure 3.3.-2 indicates that the RCS aboard the 218 Lab will be switched out, while the RCS aboard the 219 Lab will be switched into operation after docking of the two Labs. All modes of operations and capabilities of the system remain unaltered. Also Flight 218 will accept inputs from a navigation experiment package.

3.3.3 Configuration III

This system adds an automatic earth-pointing capability to the basic configuration I. This is accomplished by adding a horizon scanner unit, plus star trackers and a data processor. The horizon scanners will be utilized for establishing local vertical, while the star trackers will be used for azimuth heading controls. The outputs from the scanner and trackers will be processed in the data processor to perform the required coordinate conversions to generate vehicle body control signals. The output from the processor will be fed directly into the ATCA and thence to the RCS for commanding the required attitude control. An attitude lock-on display will be provided to indicate deviations from the established star tracker and scanner positions. The block diagram for this configuration is shown in Figure 3.3.-3 and will be employed for Flights 215 and 518D. Flight 215 is also a separated artificial "g" flight, but only for short durations. Attitude sensor inputs can also be routed to the CSM SCS when the SM RCS is being used.

3.3.4 Configuration IV

This system is designed to perform functionally like configuration III above, and is identical to it in all respects - with the exception that an earth tracker, rather than horizon scanner, will be utilized to establish local vertical. This change in equipment is necessitated by the synchronous orbit planned for Flight 516D, for which this configuration is intended. The data processor will be programmed to operate with the star trackers, to provide an inertially stabilized mode of control. The block diagram of the configuration is shown in Figure 3.3.-4.

3.3.5 Configuration V

This system is composed of configuration I, plus both the star tracker and horizon scanner-processor mechanization is programmed to provide inertial and earth-oriented stabilization modes of control. The configuration is shown in the block diagram in Figure 3.3.-5, and is planned for use on Flights 523 C & D, and 229. The flight requires independent maneuvering for rendezvous, in addition to long-term inertial pointing, and is capable of controlling the RCS on Flight 229. Attitude sensor inputs can also be used to interface with the CSM SCS when in use.

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3.3.6 Configuration VI

This system is intended for use on Flight 509, which requires separation from, and docking with an unmanned LEM Lab by the CSM. A communication Antenna/Command and Coupler Electronics mechanization will be provided to operate with the ATCA/Rate Gyro/RCS mechanization. In this manner the CSM may command tight deadband control of the LEM Lab during docking while separated. Calculations show that from a weight/power point of view this arrangement is superior to carrying additional RCS propellant on board and operating the LEM Lab in the tight deadband limits for the duration of the experiment. This flight also requires periods of both inertial and earth orientation using either the CSM or LEM Lab Stabilization System. Star trackers and an earth tracker system have been added to provide long term attitude reference. A block diagram of this configuration is shown in Figure 3.3.-6.

3.3.7 Configuration VII

This system shown in Figure 3.3.-7, is designed to utilize the CSM control system for an inertial pointing mode of operation. The basic attitude control is desired from the LEM Lab, this is accomplished by providing an attitude lock-on display, attitude rate error needles, and a hand controller on the LEM Lab panel. An earth tracker has been added to provide local vertical for synchronous orbital Flights 516C and 521 C & D. A horizon scanner performs the same function for Flight 518C. DOD 1 C&D carries only star trackers.

3.3.8 Configuration VIII

This system is also designed to utilize the CSM control system for an earth-pointing mode of operation in which star trackers and horizon scanner are mounted on the LEM Lab, but for which no manual control or pilot monitoring is required from the Lab. This configuration is shown in the block-diagram in Figure 3.3.-8, and is planned for Flights DOD 2 C&D, DOD 3 C&D, DOD 4, DOD 5, DOD 6, 513, 507 and 211 C&D.

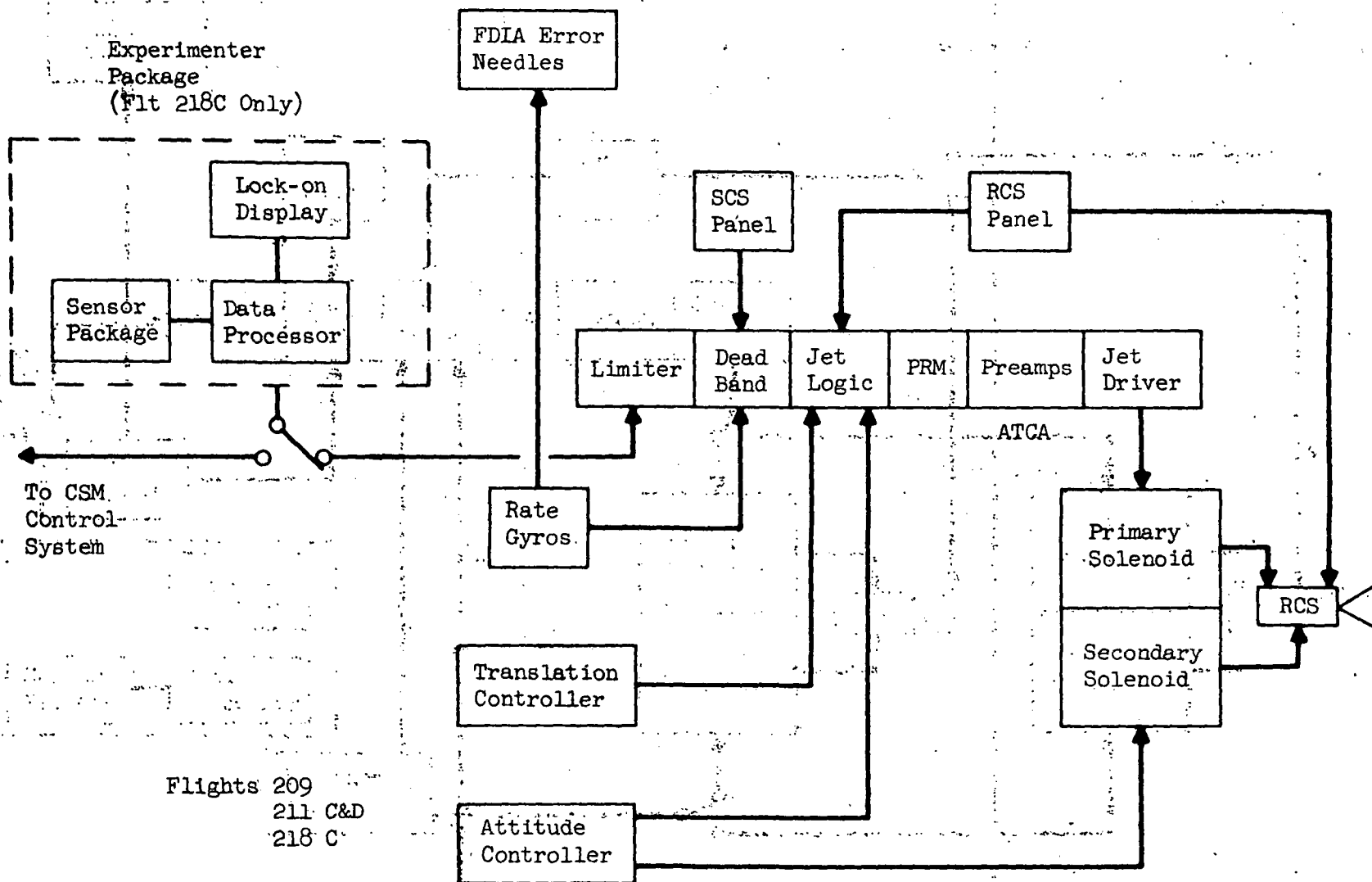


Fig. 3.3-1 LEM Lab Integrated Flight Control System
Configuration I

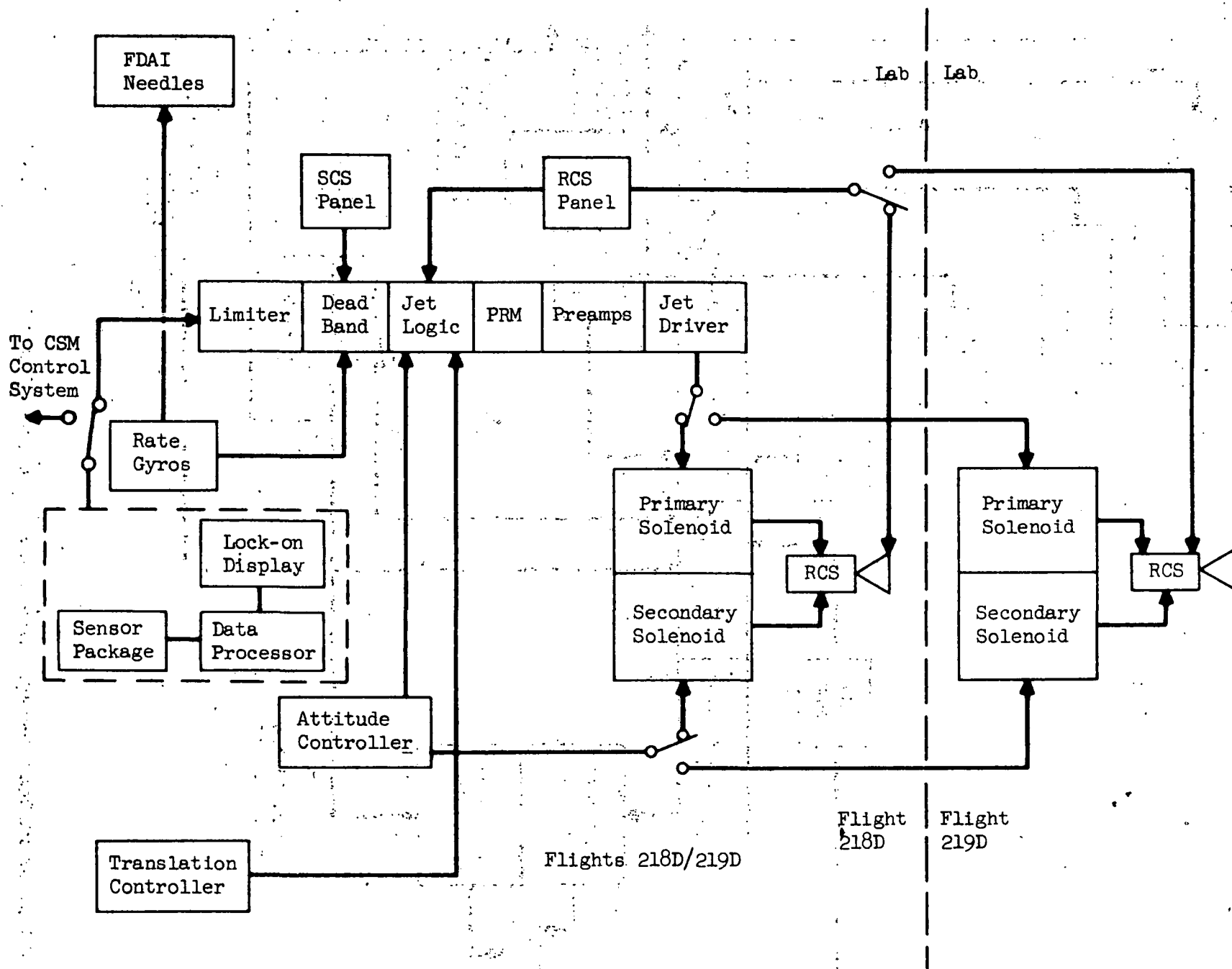


Fig. 3.3-2 LEM Lab Integrated Flight Control System Configuration II

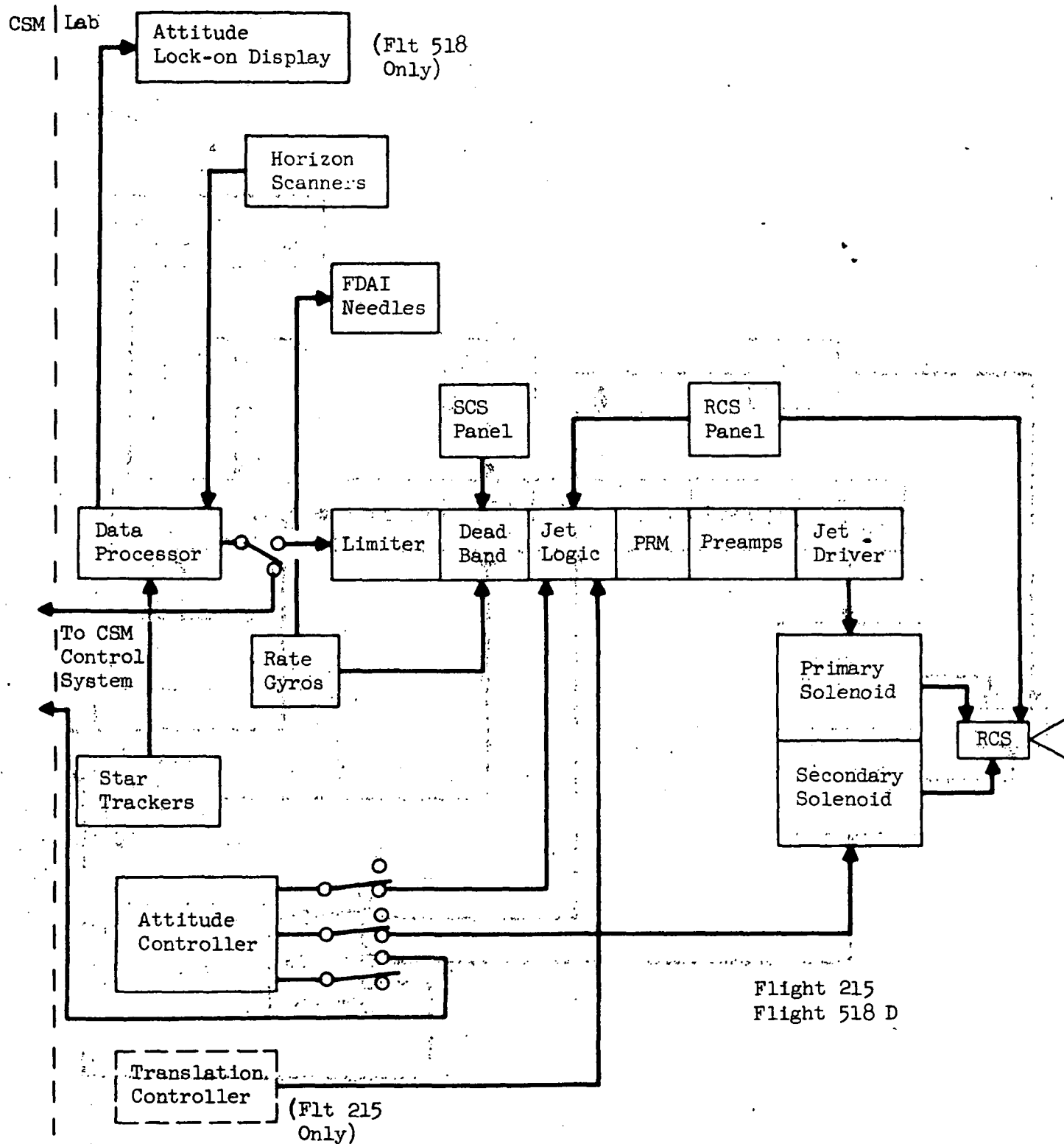


Fig. 3.3-3 LEM Lab Integrated Flight Control System Configuration III

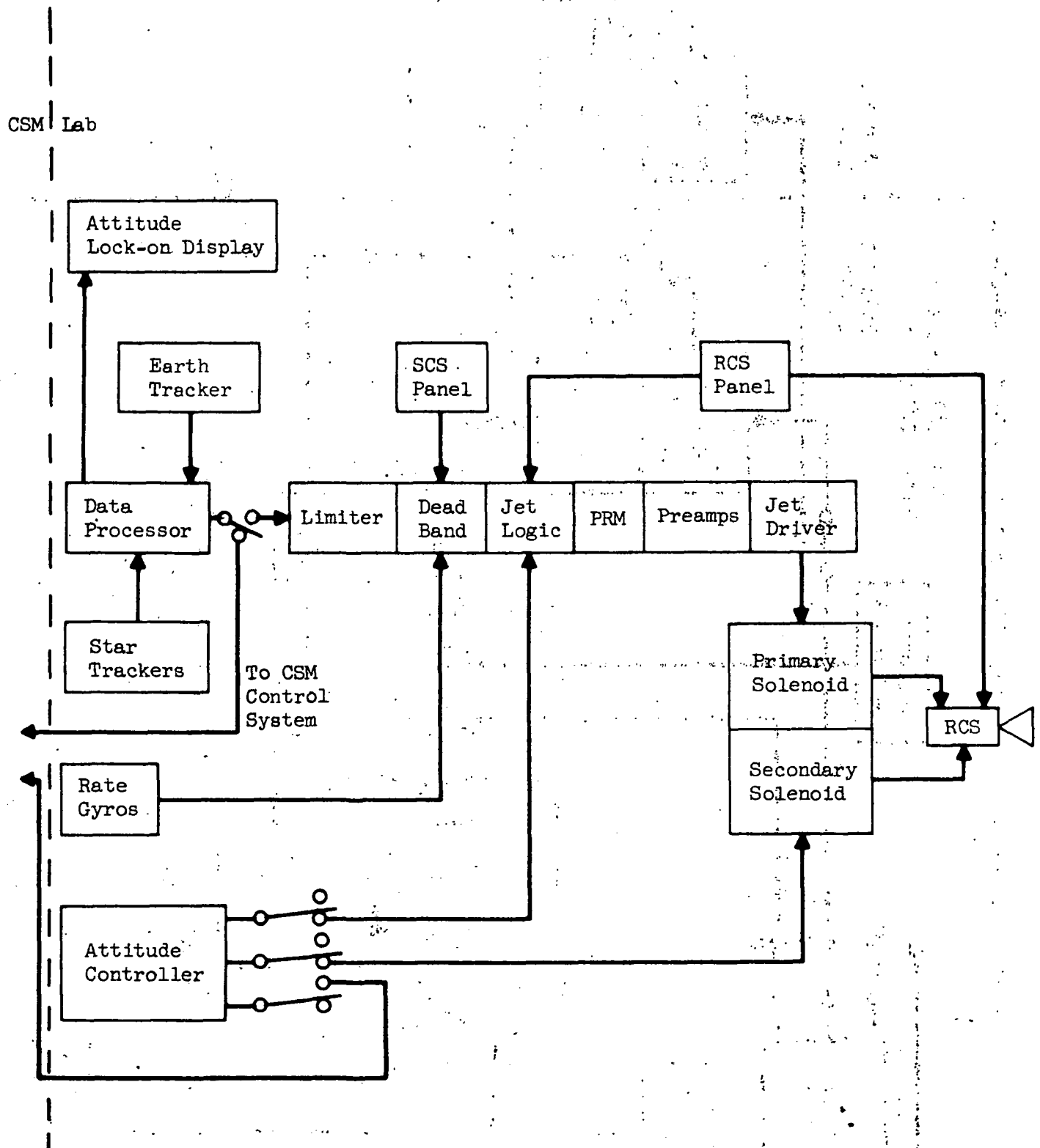
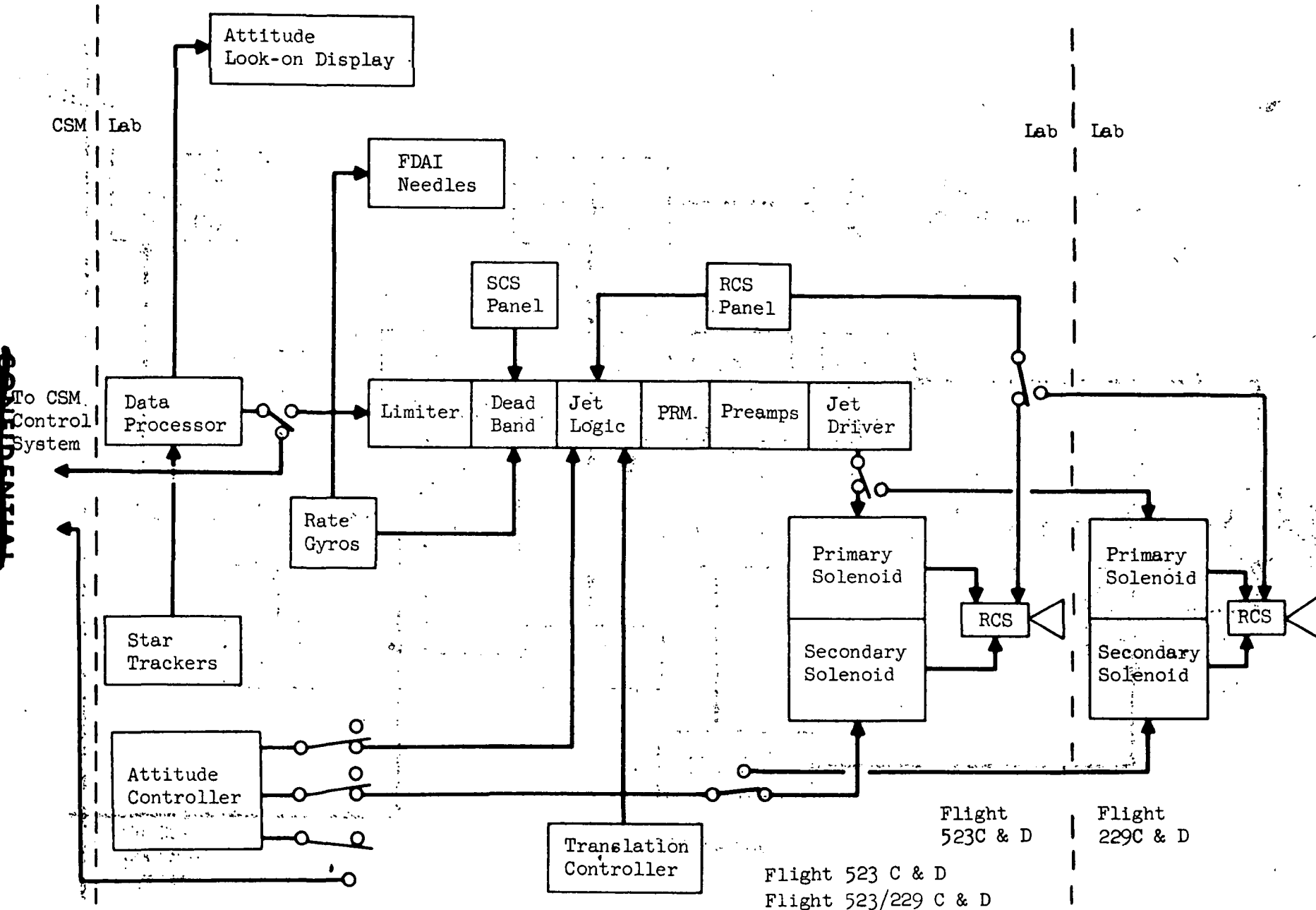


Fig. 3.3-4 LEM Lab Integrated Flight Control System Configuration IV

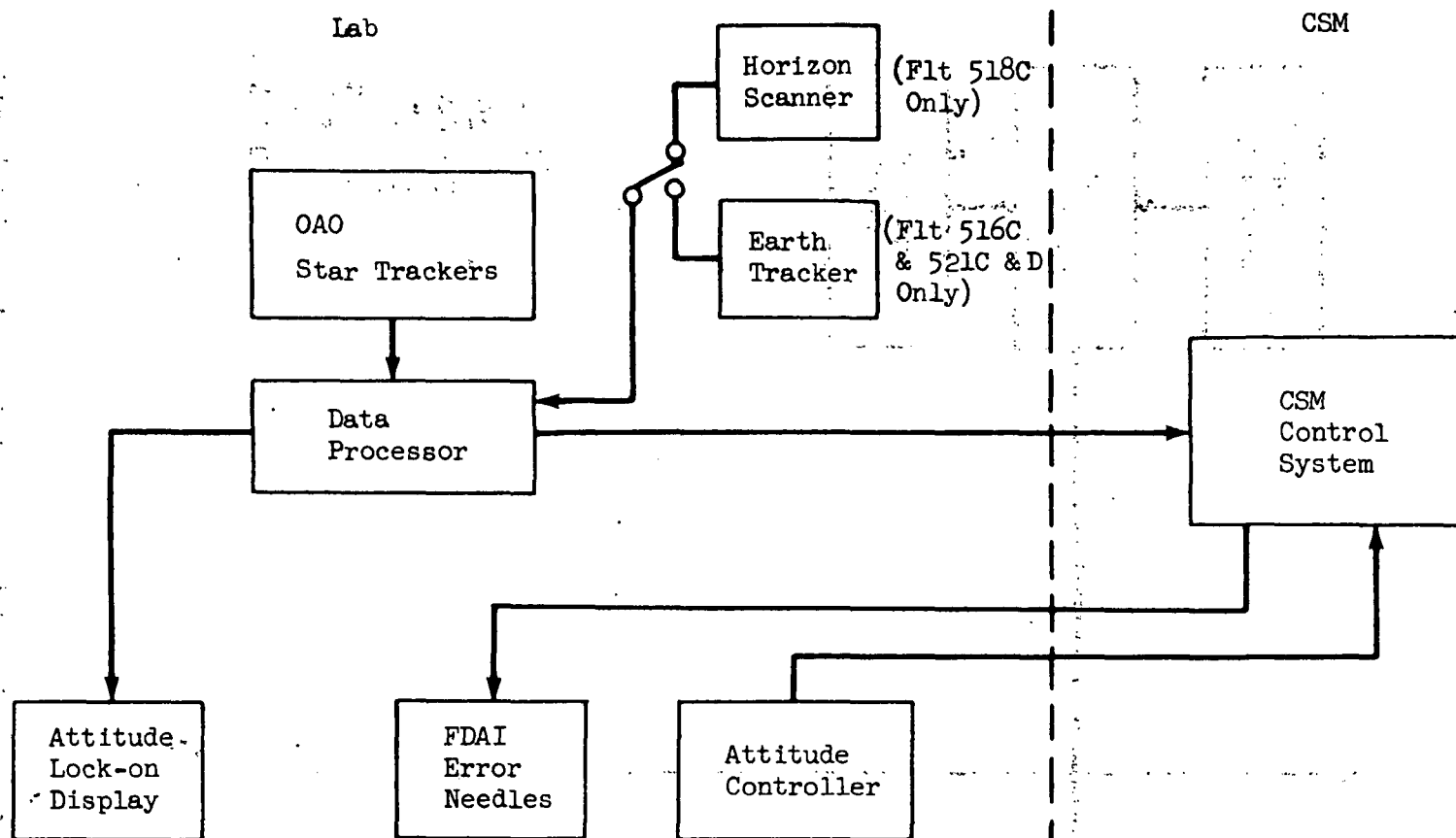
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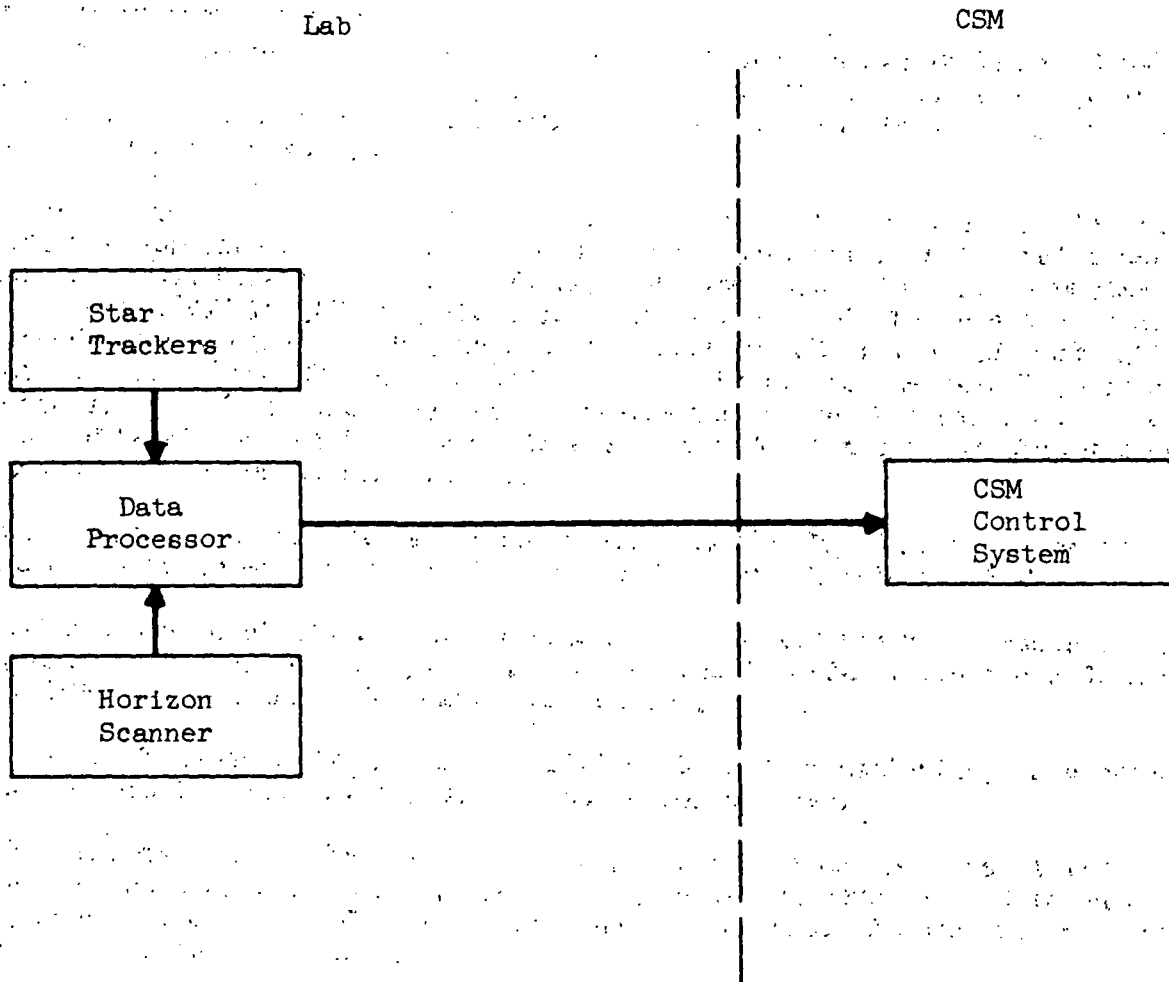


Flight No.
DOD - 1 C & D

516 C

518C
521C & D

Fig. 3.3-7 LEM Lab Integrated Flight Control System Configuration VII



Flight No.

DOD - 2C & D
 DOD - 3C & D
 DOD - 4
 DOD - 5
 DOD - 6
 513
 507
 211 C & D

Fig. 3.3-8 LEM Lab Integrated Flight Control System Configuration VIII

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3.4 DATA HANDLING AND COMMUNICATIONS

3.4.1 General Description

The basic function of the Lab data handling subsystem, shown in Figure 3.4-1 is to control the generation, storage, processing and routing of both the experimental data and the LEM status data. The communication function is normally handled by the CM communication subsystem.

The system shown consists of the experiment sensors, status determination sensors, and their respective signal conditioners. The experiment signal conditioners are connected to the Experiment Select Panel; the panel controls those experiments applied to the PCM-TEA (Timing Electronic Assembly). The TEA generates timing and reference pulses, converts the experimental data to PCM (pulse code modulation) and routes the serialized NRZ data through the Mode Switch to the CM-PMP (promodulation processor). The Mode Switch controls time-sharing of the Lab experimental data from the PCM-TEA and the Lab status data with the CM status data; thus a single PMP and communication subsystem is used.

A Data Control Panel allows selection of voice and/or PCM data to be recorded on the Data Storage Recorder. Playback is via the Mode Switch into the PMP.

The Audio Center controls the Lab intercom function and the voice recorder (Push-to-Talk, or, voice operated relay), and provides the interface, via the Mode Switch, to the PMP intercom and VHF/S-Band transceivers.

A slow scan TV camera, for use with real-time S-band transmission, is provided for Biomedical and Behavioral experiment observation by MSFN.

All Instrumentation and Communication equipment in the CM are essentially the same as for the basic Apollo mission. The high gain S-band antenna is deleted except for synchronous orbits. The TV is contained in the Lab. A Mode Switch is added to control data flow into the PMP.

On flights where the Lab is always attached to the CM, the LEM uses a minimum of its standard communication and instrumentation equipment: audio center/intercom (present LEM), data control panel (new), PCM Timing Electronic Assembly (modified LEM), sensors and signal conditioning for Lab status data (modified LEM), and sensors and signal conditioning for experimental data (new and peculiar to each experiment.)

3.4.2 Communication Analysis

The existing CM communication subsystem is used to transfer the CM status, Lab status and Lab experimental data to the Earth as well as to provide communication with an extravehicular astronaut. The following table shows the equipment combinations and system margins for transmitting 51.2 K bits/sec. of data.

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<u>ORBIT</u>	<u>OUTPUT POWER</u>	<u>VEHICLE ANTENNA</u>	<u>GROUND ANTENNA</u>	<u>SYSTEM MARGIN</u>
Sync.	20 w	Omni	30 ft	neg.
	20 w	Omni	85 ft	+ 1.6
	20 w	Hi Gain	30 ft	+21
	20 w	Hi Gain	85 ft	+29.6
	5 w	Hi Gain	30 ft	+15
	5 w	Hi Gain	85 ft	+23.5
	250 mw	Hi Gain	30 ft	+ 2
	250 mw	Hi Gain	85 ft	+10.5
200 n. mi.	20 w	Omni	30 ft	+21
	5 w	Omni	30 ft	+15
	250 mw	Omni	30 ft	+ 2
160 n. mi.	20 w	Omni	30 ft	+26.5
	5 w	Omni	30 ft	+20.5
	250 mw	Omni	30 ft	+ 7.5

The amount of available communication time per orbit is 100 percent of the orbital time for synchronous orbit, or variable, as shown in Figures 3.4-2 to 3.4-6 for low altitude orbits. The figures consider only those NASA ground sites which have S-band communication capability. These are:

<u>STATION</u>	<u>DEG LATITUDE</u>	<u>DEG LONGITUDE</u>
Madagascar	18	48
Carnarvon	25	114
Canberra	35	148
Guam	12	145
Kauai	22	200
Guaymas	28	248
Cape Kennedy	28	279
Wallops Island	37	284
Bermuda	33	297
Canary Islands	28	344
Madrid	37	355

The communication times shown in the figures are for 4-day periods. These are considered representative for the total mission duration, and show the following average communications time for the different orbital inclinations:

<u>INCLINATION, deg</u>	<u>AVERAGE COMMUNICATIONS TIME PER ORBIT, min</u>
28.5	21.5
50	16.5
81.5	10.5
90	10.5
97 (-83)	10.5

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Of this average communication time, it is assumed that four minutes per orbit will be allocated for CSM transmissions. The remainder of the time is available for experiment data transmission.

Corresponding data for communication of the DOD flights with DOD ground sites are shown in Appendix B of Volume II.

Table 3.4-1 lists communication requirements for the CM and the Lab. As shown, all flights require CM S-Band voice (two way), 51.2 K bits/sec telemetry, video, up-data and ranging capability as well as VHF voice (two way) and biomedical (receive) capability. VHF data transceive function is required on flights such as 509 where the Lab is separated from the CM; here, Command transmittal for vehicle control is required for docking, and reception of low bit rate data is required for status evaluation. Another requirement is back-up to an umbilical connection as in Flight 215. LEM Lab requirements are minimal except during separated phases of the mission. When separation is involved, as in Flight 509, a low bit rate data capability on the S-band is desirable for Earth evaluation of status data, an up-data link on S-band is desirable for back-up vehicle control, the S-band ranging function is required to operate in conjunction with the CM. In flights where an umbilical is involved, the preceding Lab S-band and VHF functions are required as back-up as well as a voice capability on both frequency bands.

Unlisted items are those associated with AMU (astronaut maneuvering unit) and RMU (remote maneuvering unit) operation. These units are considered experiments, although they are used on many missions. The AMU presently is compatible with the CM in terms of the voice function; however, a 433 mc receiver and antenna function is required on the CM, to provide the telemetry capability. To be compatible with RMU operation, a command transmitter, communication transceiver, telemetry receiver, video receiver and radar transceiver are required on the Lab, in addition to associated display and control equipment and antenna complex.

3.4.3 Data Management Analysis

Data storage is required to provide experiment data collection continuity during missions (non-synchronous orbit) that have data transmission limitations both in frequency and duration. The proposed Apollo-type PCM and magnetic tape recording equipment fulfill this requirement.

This PCM equipment combines analog and digital inputs into a serial NRZ digital data train at either of two bit rates: 1.6 or 51.2 K bits/sec. Problems of patching and programming the experiment data into the PCM unit have no simple solution, due to the diversity of experiment data requirements. Assumption has been made that each experiment will fit into either of the two bit rates. Definition of the proper technique will require a detailed analysis of each mission.

The magnetic tape recorder impresses 14 data channels (five digital, nine analog) on a 2250 foot length of 1-inch wide tape. Digital data at 51.2 K bits/sec (PCM-NRZ) is both recorded and played back at 15 inches/sec for a period of approximately 30 minutes per reel of tape. The 1.6 K bits/sec data is recorded at 3.75 inches/sec (115 min/reel), and played back at 120 inches/sec (4 min/reel) at the higher data rate of 51.2 K bits/sec. The requirements for analog recording (except voice) are not defined.

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The basic data management system will be used for all flights, except, as noted below:

- For flights requiring separated and manned Lab operations, the recording and/or transmission of status data must be provided on the Lab.
- DOD flights 1-3 will include the DOD proposed Experiment Information System (EIS). As one of its functions the EIS will replace the PCM equipment shown on Fig. 3.4-1.

An analysis of the basic system discussed above was performed for each flight. Results are in terms of the quantity of data, if any, that cannot be transmitted because of transmission time limitations, and which must, therefore, be returned in the form of reels of tape. (It has been assumed that as much data as possible will be transmitted.)

Analysis results appear on Table 4-1 in Volume 1 for each flight. Data rates and average daily data recording times were estimated by experiment. According to its data rate, an experiment was assumed recorded at either 1.6 or 51.2 K bits/sec. The total average daily recording times at these two data rates was computed, from which the daily data transmission time requirement was defined. Comparison of required and available data transmission times yielded any excess of stored data which would require physical return.

The Apollo data handling equipment is adequate for most flights. However, analysis of tape return requirements indicates that for some of the flights a different or modified system may be desirable. Among the possible alternate approaches the following should be considered:

- Change the record speeds to achieve higher bit packing densities and a corresponding decrease in tape return weight.
- Modify the PCM and tape recorder to accommodate a third bit rate, (i.e., 6.4 K bits/sec), with the corresponding time compressed playback for a greater percentage of data.
- Modify the system completely to provide optimum playback time compression (by the use of asynchronous techniques - i.e., an incremental recorder, etc.)

Data transmission times, using the latter optimum system, have also been computed for each flight, (see Tables 4.1, Vol. 1). Calculations were based on the assumption that data from each experiment could be recorded at its estimated bit rate (plus 50 percent for format, etc.) and played back at 51.2 K bits/sec. - thus providing optimum playback time compression.

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3.5-4

In either Concept A or B, improvement in the operating life of the propellant tank bladders may be required. The permeability of the Teflon bladders by N_2O_4 , which is relatively high, may be unacceptable for long space-storage requirements, which may be as much as 135 days for Flight 523. Qualification tests should be run on the bladders for this longer storage time.

Requalification of the thrusters has been discussed above. Endurance and mission profile demonstration tests will be required for the complete system.

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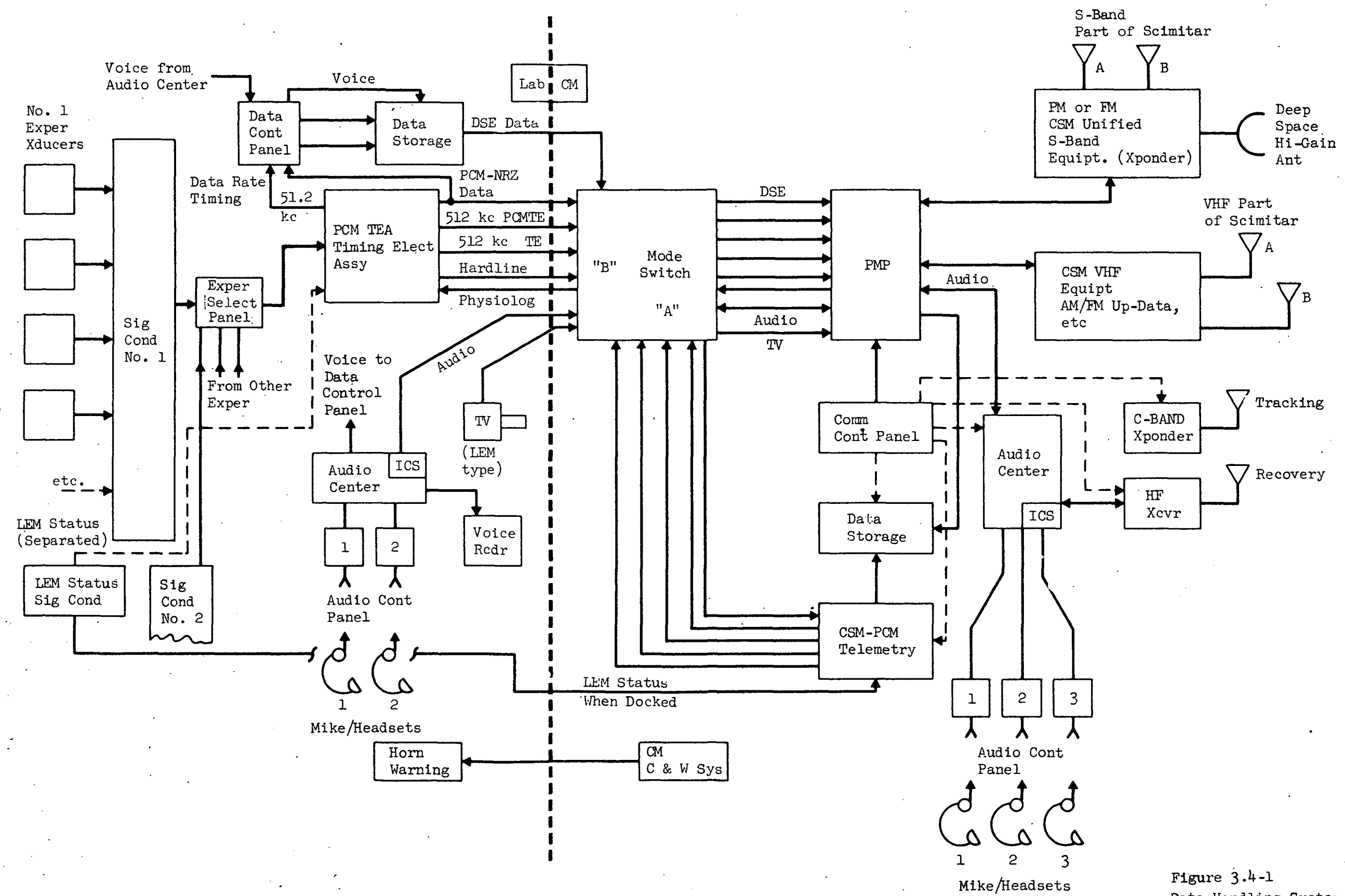
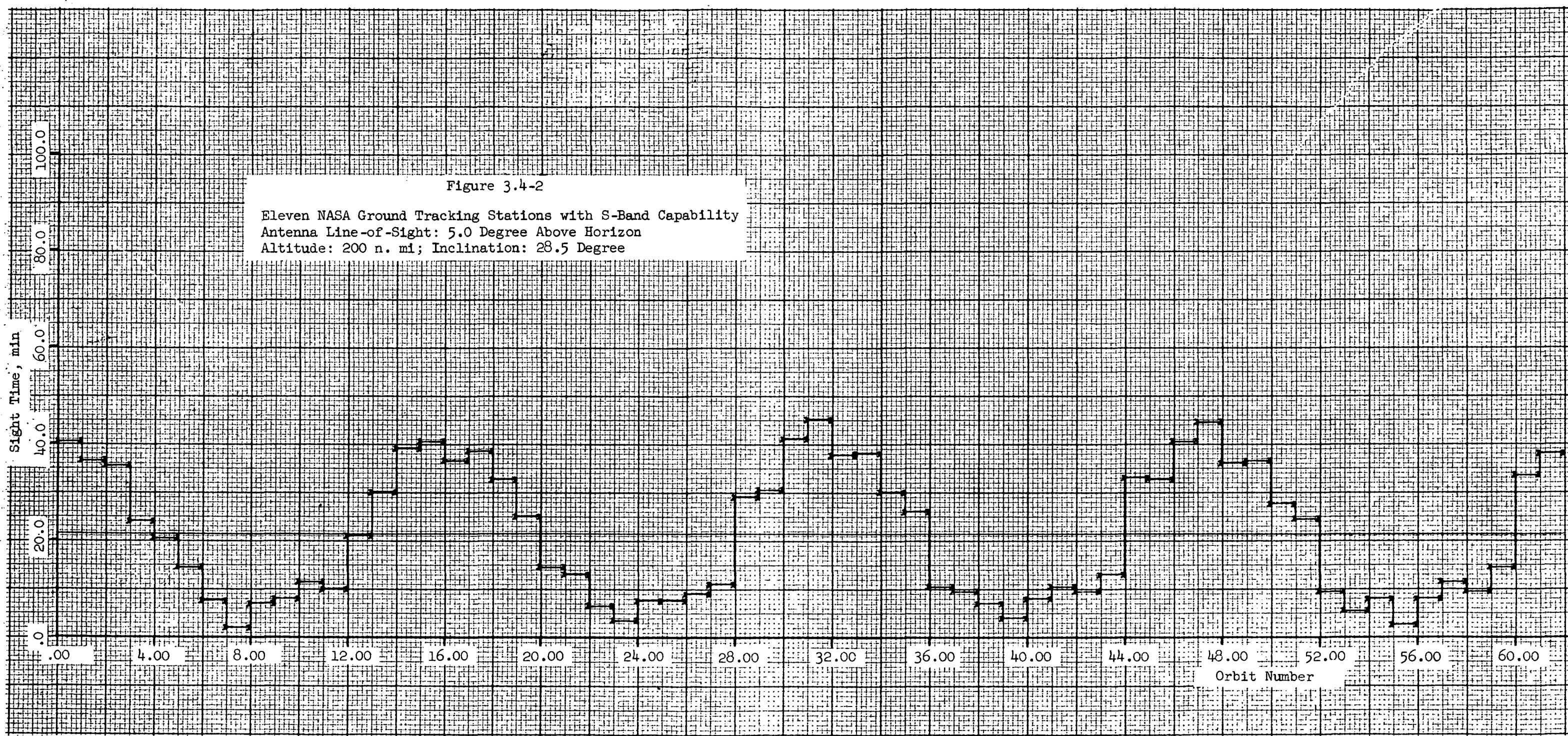
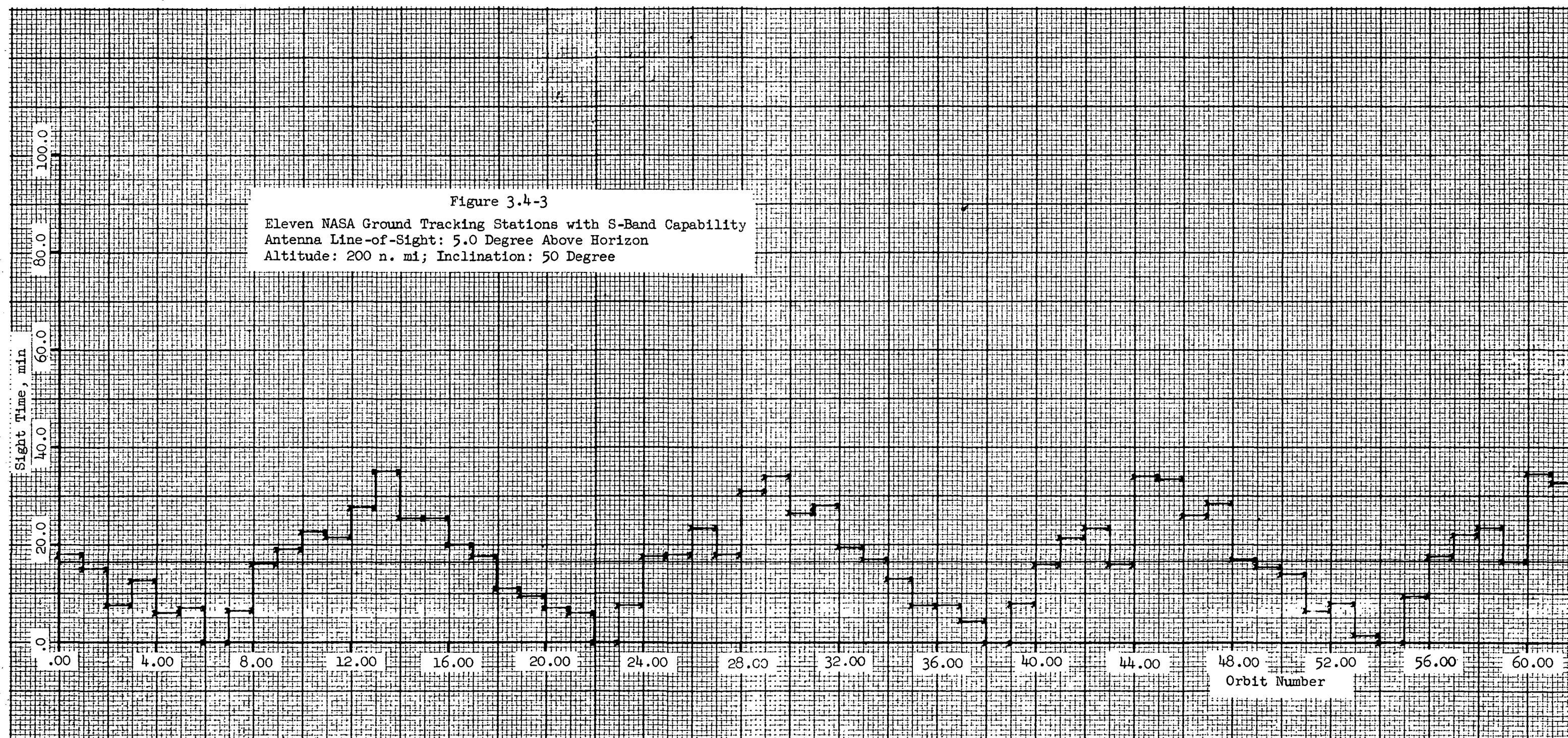
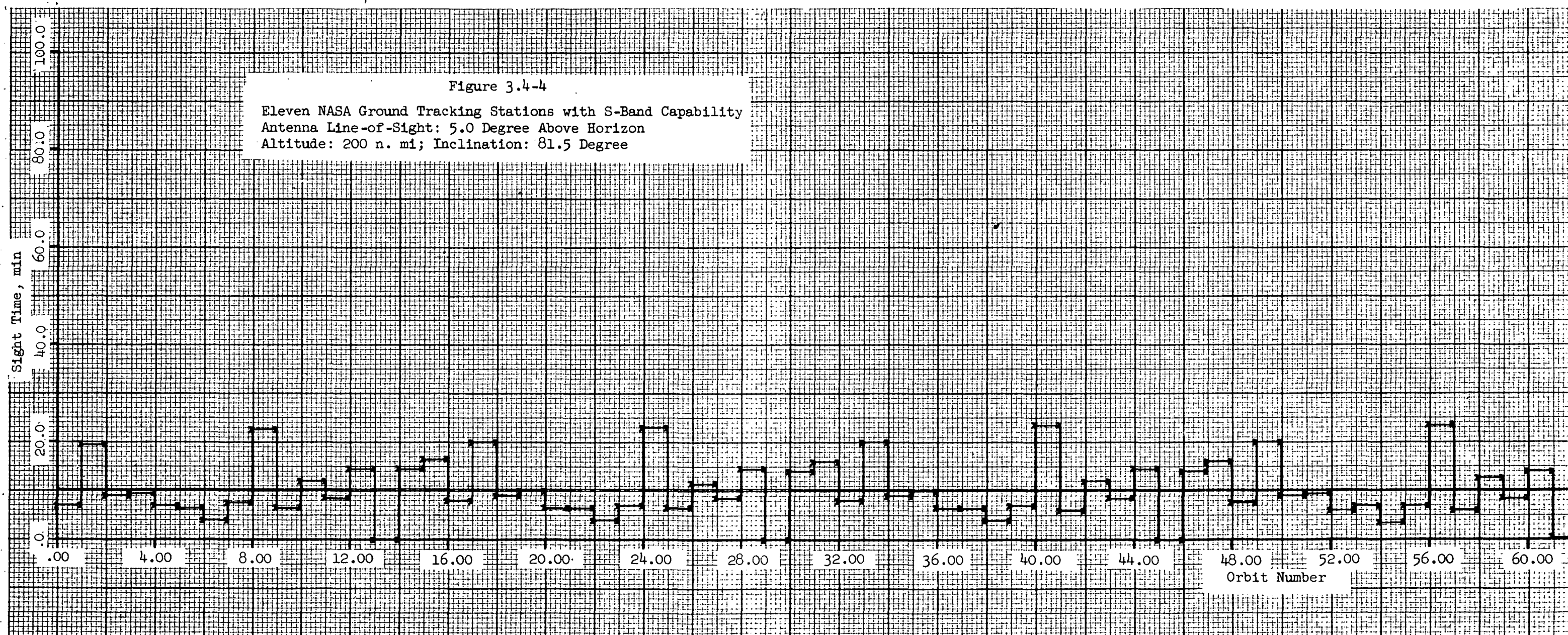
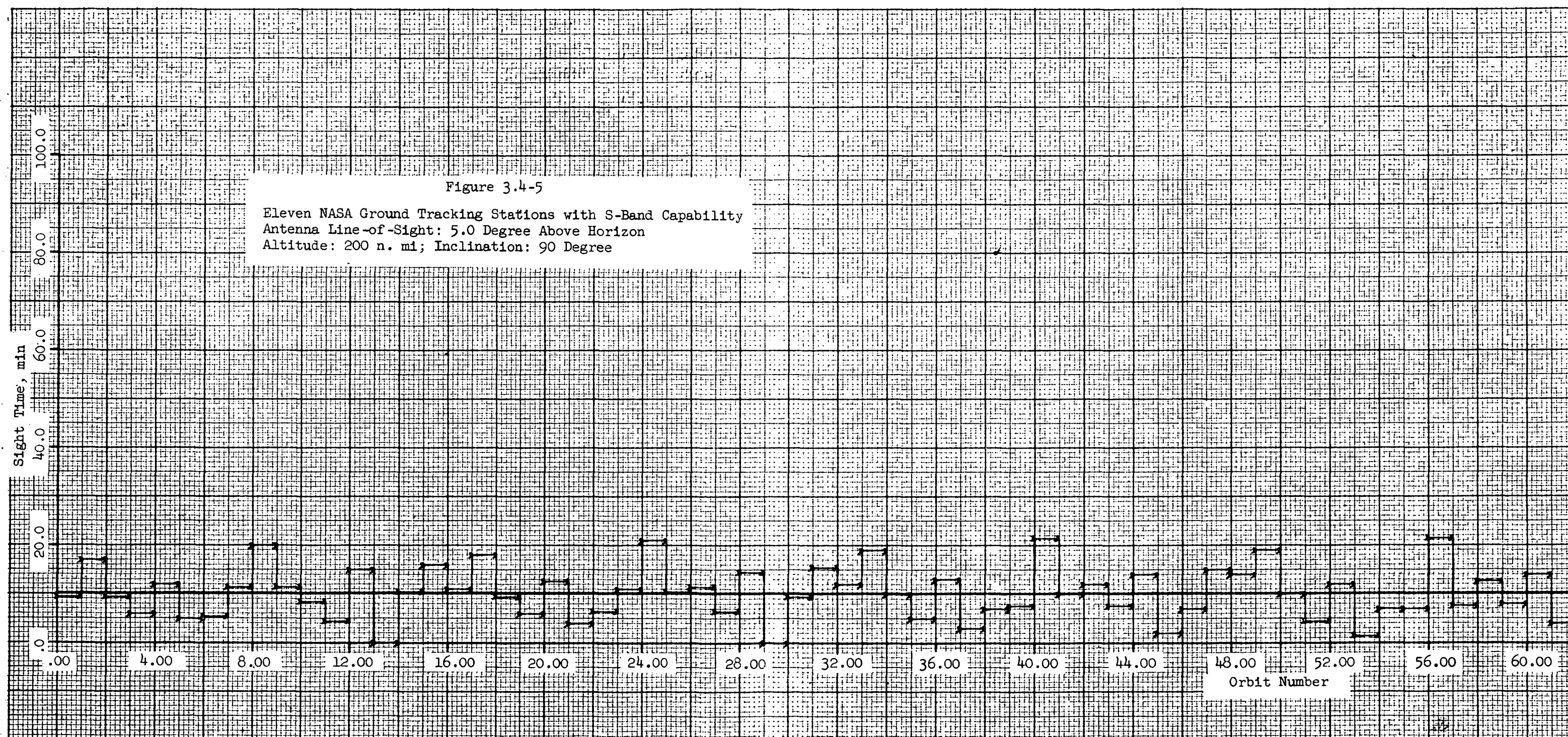


Figure 3.4-1
Data Handling System









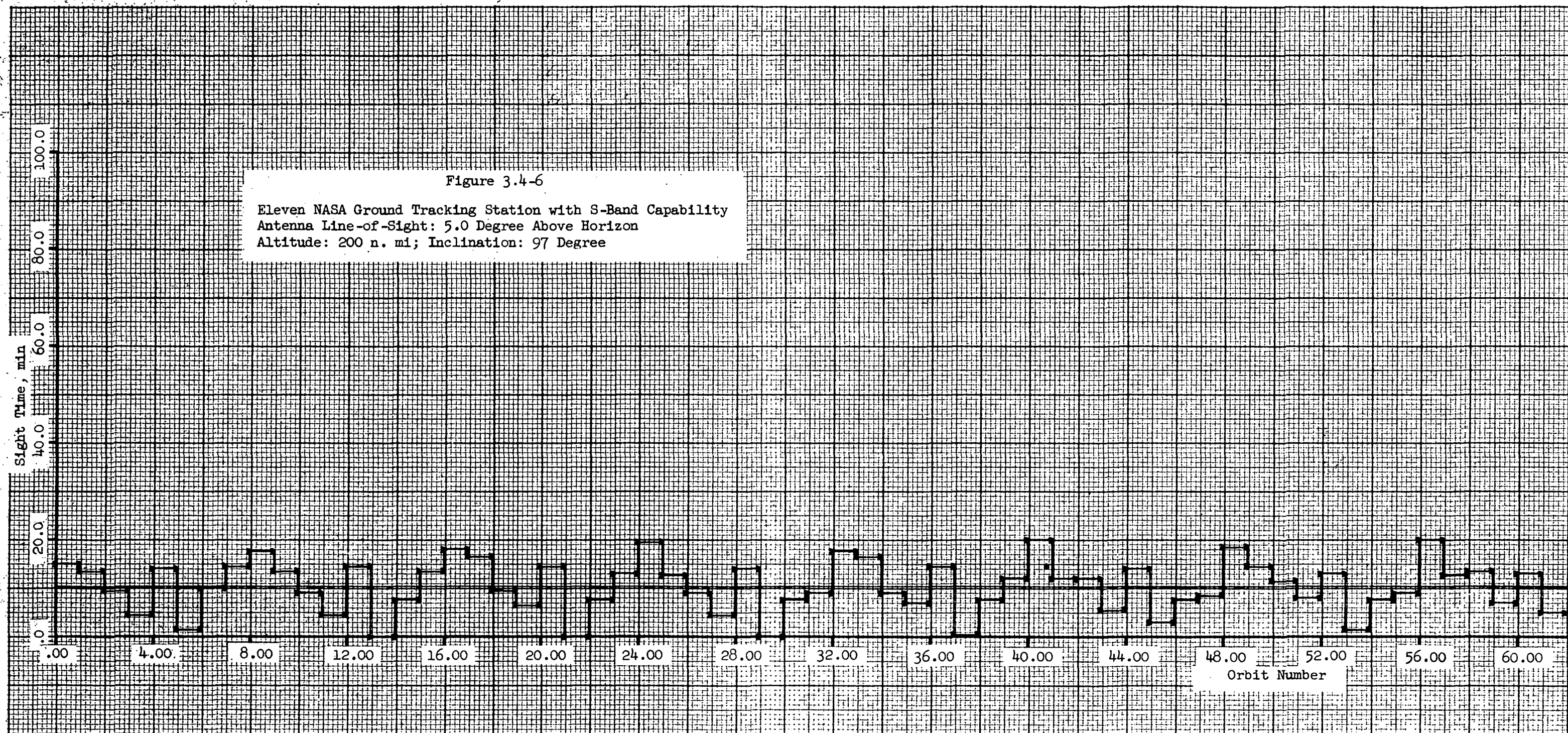


Table 3.4-1
SUMMARY OF COMMUNICATION REQUIREMENTS

		CM S-BAND					CM VHF		LAB S-BAND							
Flt.	Alt.	Voice	51.2 Kbps	TV	UpData	Ranging	Voice/Bio	Data R/T	Voice	1.6 Kbps	51.2 Kbps	TV	UpData	Ranging	Data R/T	Voice
209	200	X	X	X	X	X	X									
211	200	X	X	X	X	X	X									
507	200	X	X	X	X	X	X									
509	Sync	X	X	X	X	X	X	X X		X			X	X	X X	
215	200	X	X	X	X	X	X	X*X*	X*	X*			X*	X*	X*X*	X*
513	200	X	X	X	X	X	X									
218	200	X	X	X	X	X	X								X X	
219	200	X	X	X	X	X	X	X X							X X	X
221	200	X	X	X	X	X	X		X*	X*			X*	X*		
516	Sync	X	X	X	X	X	X									
518	200	X	X	X	X	X	X									
521	Sync	X	X	X	X	X	X									
523	200	X	X	X	X	X	X								X X	
229	200	X	X	X	X	X	X									
230	200	X	X	X	X	X	X	X X								

* Back-Up

3.5 REACTION CONTROL SYSTEM

3.5.1 Propellant Requirements

The reaction control system provides the torque for all attitude changes, attitude stabilization, and in some cases provides the thrust for small velocity changes. The service module RCS is used to fulfill these requirements up to an assumed maximum capacity of 2280 lbs. of propellant for "C" configurations and 790 lbs. for "D" configurations except in cases where precluded by vehicle separation. If needed, additional propellants are carried on the LEM and are burned in the independent LEM Reaction Control System.

Propellant requirements have been computed for LEM and SM systems for each flight based upon the mission plan and experiment requirements. System parameters used in this study are shown in the following:

REACTION CONTROL SYSTEM PARAMETERS

Jet thrust - 100 lbs.

Minimum impulse bit during pulsing - 0.6 lb. - sec.

Jet arm (distance between jets) -

13.4 ft. on SM

11.0 ft. on LEM

Attitude Deadband:

$\pm 5^\circ$ and $\pm 0.5^\circ$, or $\pm 5^\circ$ and $\pm 0.2^\circ$ (optional)

Limit Cycle and Rotational Maneuvers

- 2 jets fire in couple

Specific Impulse:

a. Isp = 100 seconds for minimum impulse bit pulsing; eg.

Limit Cycle, Celestial Pointing and Altitude Maintenance

b. Isp = 240 seconds for Rotational Maneuvers

c. Isp = 263 seconds for Translational Maneuvers

Propellant requirements for earth-oriented stabilization assumed the long axis horizontal and in the orbital plane, and therefore were based only on maintaining attitude within deadband requirements. Propellant requirements for inertial stabilization were based primarily on estimates of disturbances due to magnetic and solar pressure effects are believed small and were therefore neglected.

Table 3.5-1 shows the results of the study. For each flight, the principal operating requirements are given together with the propellants required. In general, unless otherwise specified, a limit cycle of $\pm 5^\circ$ was assumed together with an

earth pointing orientation. This insures proper orientation for earth communication and gives minimum orientation fuel consumption.

For altitude maintenance, an Isp of 100 seconds was used, assuming a large number of small pulses rather than a few longer pulses with higher Isp. Reorientations to line up thrusters with the flight path were assumed unnecessary. For rotational maneuvers, an angular rate of 0.4 deg/sec. was assumed unless more specific requirements could be obtained from the experiment requirements. This is an important consideration in flights 219, 516, 523, 229 and 230 and better definition of the angular rates to be used could appreciably change propellant requirements.

A preliminary calculation showed the gravity gradient torque due to celestial pointing would be resisted by pulsing couples, and the corresponding Isp of 100 seconds was used. For lack of more specific data, an arbitrary orientation of the vehicle with respect to the orbital plane was assumed during the celestial pointing periods. The resultant fuel consumption, therefore, is probably conservative. If final experiment arrangement permits orientation of the long axis in the orbital plane, the final consumption will be reduced by as much as 40 per cent.

Docking requirements for the active vehicle were estimated by assuming an equivalent linear ΔV of 25 ft/sec., except for the docking experiments of Flight 509 where 35 ft/sec. was assumed.

Total propellants for each flight are shown in Table 3.5-1 for both "C" and "C" configurations. Propellants for ascent and pre-entry phases were taken from North American Aircraft documents, and a 10% reserve was added as shown.

In some instances, Flights 219, 516 D, 518D, 523 and 229 and 230, the requirements exceed the SM tankage limitations stated above. In these cases a Special Propellant Distribution is shown. Arrows indicate the manner in which the distribution is made, eg: in Flight 219 "D" the excess over 790 lbs. is stored partially in the #219 LEM which therefore has an RCS system added, and partially in the #218 LEM which joins Flight 219 after completion of Flight 218.

The greatest amount of propellant is 4630 lbs., used in the Flight 523 "D" LEM. Flight 229 "D" LEM requires 3700 lbs. and several flights require 2280 lbs. As shown in the table, Flight 229 "D" LEM requires no propellant in itself, but supplies the excess over the SM storage capabilities. Most of the Flight 523 "D" LEM propellant is used to supplement SM capabilities in the 523 and 230 flights; note that Flight 523 LEM joins Flight 229 and then Flight 230, being in orbit 135 days. The propellant requirements of these flights are due primarily to the celestial pointing and rotational maneuvers demanded by the 0700 experiments which are on all three flights. The requirements are also large because of the large moments of inertia of Flights 229 and 230.

Additional studies will be required to evaluate the effects of O/F ratio on tankage requirements. When the percentage of propellant used during pulsing is large, the O/F ratio will deviate appreciably from the steady state value of 2:1 and may cause a change in the number of oxidizer and fuel tanks carried.

3.5.2 Angular Rate

The angular velocity that will be experienced during limit cycle operation is also shown in Table 3.5-1. Rates are given for both pitch and roll and were computed for the case where the rates are equal in either direction of the cycle. Actually, limit cycle rates are a function of the initial angular disturbance which is random. Thus the rates could vary from practically zero in one direction to nearly twice the values shown in the opposite direction.

3.5.3 Thruster Firing Time

For each flight, an estimate was made of the number of firings and the total firing duration of the most used thruster. These values are shown in Table 3.5-2 for both the SM and LEM, using the special propellant distribution from Table 3.5-1. Also shown in Table 3.5-2 is the minimum time the RCS system must be pressurized to meet the flight requirements. The requirement of 1055 seconds firing time and 108,000 pulses is considerably more than the present requirements of 1000 seconds and 10,000 pulses. However, reliability studies which are discussed more fully in 4.2 indicate the present RCS system components should be adequate for these requirements, although qualification tests will be required.

3.5.4 RCS Arrangement for Increased Propellants

The baseline Reaction Control System for the AES Lab will be the present LEM system with minor modifications to make it applicable for this vehicle. The only area which requires changes of any magnitude is the arrangement of a series of tanks to make-up the additional propellant required for the extended mission. Two system concepts are presented in Figures 3.5-1 and 3.5-2. A brief description of these concepts is presented below:

Concept A - (Figure 3.5-1) - uses multiples of the present LEM RCS modules. In this concept new design and fabrication of components would not be required. Additional support structure, a simple tie-in to the propellant manifolds and minor modifications to the control panel would be required. This concept represents the minimum requirement for component requalification in that the sequential operation of each module will not increase the life requirements of the components with exception of the thruster assemblies. This is reflected in the Design and Development Cost Program.

Concept B - (Figure 3.5-2) - uses multiples of the present LEM RCS helium and propellant tanks with a single set of pressure regulation components per system. In this concept, new design and fabrication of components also would not be required. Additional support structure and simple tie-in of the helium and propellant manifolds would be required. No modifications to the present control panel would be necessary. However, the PQGS would have to be modified to integrate the liquid levels of the additional tanks. This concept requires additional component requalification in that the life of the components must be increased to meet the extended mission requirements. However, concept B also represents the minimum weight system.

Concepts A and B represent the extremes of the possible RCS configurations. Further study may determine that a system somewhere in between these will be the optimum for the AES Lab.

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Table 3.5-1
REACTION CONTROL PROPELLANT

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3.5-5

Limit Cycle Rate, deg/sec		Roll Pitch	.0149 .0023	.0114 .0018	.0134 .0019	.0084 .0013		.0134 .0019		.0127 .0019		.0096 .0006	.0140 .0020			.0067 .0009	.0098 .0016	.0070 .0010	.0090 .00127		.0078 .0005		.0078 .0008	.0119 .0018	.0117 .0019	.0122 .0018	.0118 .0016	.0118 .0016	.0118 .0016	
Orbit Altitude		n. mi	200	200	200	19,350		200		200		200	200	200		19,350	200	19,350	200		200		200	200	200	200	200	200	200	200
Flight Duration		days	14	30	14	14		14		45		45	14	45		45	45	45	45		45		45	30	30	30	14	14	14	
Flight			209 SM	211 SM	507 SM	509 LEM SM		215 LEM SM		218 LEM SM		219 SM	513 SM	221 LEM SM		516 SM	518 SM	521 SM	523 LEM SM		229 LEM SM		230 SM	SM DOD 1	SM DOD 2	SM DOD 3	SM DOD 4	SM DOD 5	SM DOD 6	
Limit Cycle ±0.2°		Days			7				7							22.5						17	17		6.45	5.05			1	
		lb			311				315								802 (D) 753					370 (D) 382	402 (D) 412		277 (D) 255	209 (D) 202			39	
Limit Cycle ±0.5°		Days		3		4	0.5			0.25	0.8	0.4	6.2			23.4		7		13.8			3.4							
		lb		49 (D) 47		125	9			30 (D) 40	14 (D) 14	6 (D) 6	119			225 (D) 206		70 (D) 65		173 (D) 164			55 (D) 51							
Limit Cycle ±5.0°		Days	14	27	7	10			7		44.2	45	7.8			21.6	22.5	38		25.8		28	28	27.4	24.7	28	14	14	13	
		lb		44 (D) 42		12	11		13		76 (D) 76	53 (D) 50	15			21 (D) 19	32 (D) 30	38 (D) 35		33 (D) 31		25 (D) 25	27 (D) 27	49 (D) 44	42 (D) 40	50 (D) 46	22	22	20	
Altitude Maintenance		Days			14				14		45	5				45														
		lb			48				48		155 (D) 155	17 (D) 17					155 (D) 155													
Rotational Maneuvers		No. @ rate, deg/sec								6@ 0.4°/s 14	300@ 0.4°/s 14					190@ 0.4°/s				118@0.2 302@ 0.4°/s		180@.16 1@ 0.8	180@ 0.16	80@ 0.4°/s	140@ 0.4°/s	120@ 0.4°/s	8@ 0.4°/s	8@ 0.4°/s		
		lb								14 (D)	2117 (D) 1996					883(D) 897				1117 (D) 1206		650 (D) 669	386(D) 400	177(D) 190	310(D) 331	270(D) 287	11	11		
Celestial Pointing		Days								0.25						18.8				25.4		17	17	3.4		0.7				
		lb								8 (D) 8						5 (D) 5				1140 (D) 1255		2360 (D) 2440	1323(D) 1390	66 (D) 70		14 (D) 15				
Rendezvous & Tracking		lb				760	20						60																	
Docking		No.					12			1	1									2	1									
		lb								29 (D) 23	64 (D) 64									100 92	63 (D) 73									
Artificial g & Centrifuge		No.						1	1					9	9															
		lb																												
Mission Propellants Ascent & Pre-Entry 10% Reserve Total Propellants		"C" Config lb	28 97 12 137*	89 97 19 205	371 97 47 515	896 - 90 986	1919 102 202 2223	169 - 17 186	551 97 65 713	63 - 6 69	337 70 41 448	241 70 221 2432	194 97 29 320	969 - 97 1066	1349 70 141 1560	1127 102 123 1352	938 97 104 1139	100 102 20 222	92 - 9 101	2729 70 280 3079		3516 70 359 3945	2299 70 230 2529	355 97 45 497	626 97 72 795	550 97 65 712	33 70 10 113	33 70 10 113	59 70 13 142	
Mission Propellants Ascent & Pre-Entry 10% Reserve Total Propellants		"D" Config lb		93 97 19 209						59 - 6 65	331 70 40 441	2263 70 23 256		Same as "C"		1134 102 124 1360	989 97 109 1195	108 102 21 231	100 - 10 110	2526 70 260 2856		3405 70 348 3823	2138 70 221 2429	347 97 44 488	629 97 73 799	543 97 64 704				
Special Propellant Distribution	"C" Config	LEM SM								300	448	0				0	1352	0		2280	2280	1130	2280							
	"D" Config	LEM SM								1140	441	1140				695	405			4630	790	3700	790							
													</																	

* AM additional 575 lb of propellant has been carried in LEM for EVA rescue only.

Corrected for Thruster.
Radii on LEM & SM

Table 3.5-2
REACTION CONTROL SYSTEM
NUMBER OF FIRINGS AND FIRING DURATION OF CRITICAL THRUSTER

Flight	Config.	209	211	507	509	215	218	219	513	221	516	518	521	523	229	230	DOD 1	DOD 2	DOD 3	DOD 4	DOD 5	DOD 6	Maximum Case
SM																							
No. of Firings	"C"	4300	6440	24,600	20,200	14,700	4590	5700	8000	7210	9110	30,000	7250	45,000	66,500	54,000	4910	12,300	10,500	2460	2460	2460	66,500
	"D"		6550				4520	1970			5280	20,800	7550	15,600	23,000	18,700	4820	12,400	10,400				23,000
Duration of Firings, sec	"C"	26	38	87	495	134	42	625	30	729	318	179	44	562	463	468	103	128	114	22	22	22	625
	"D"		39				41	217			185	124	46	195	160	162	101	129	112				217
LEM/LAB																							
No. of Firings	"C"				7,300	857	1217	-		4940	-	-		53,340	33,000	0							53,340
	"D"						3465	2780			4650	10,600		97,900	108,000	0							108,000
Duration of Firings, sec	"C"				236	87	73	-		500	-	-		508	230	0							508
	"D"						313	305			163	63		1,055	750	0							1,055
Total Time System Is Pressurized																							
LEM/LAB, Days	"C"				3	0.2	5	-		45	-			51	15	0							51
	"D"						20	20			25	15		80	40	0							80
SM, Days	"C"	14	30	14	14	14	45	45	14	45	45	45	45	45	45	45	30	30	30	14	14	14	45
	"D"		30				45	45			45	45		45	45	45	30	30	30				45

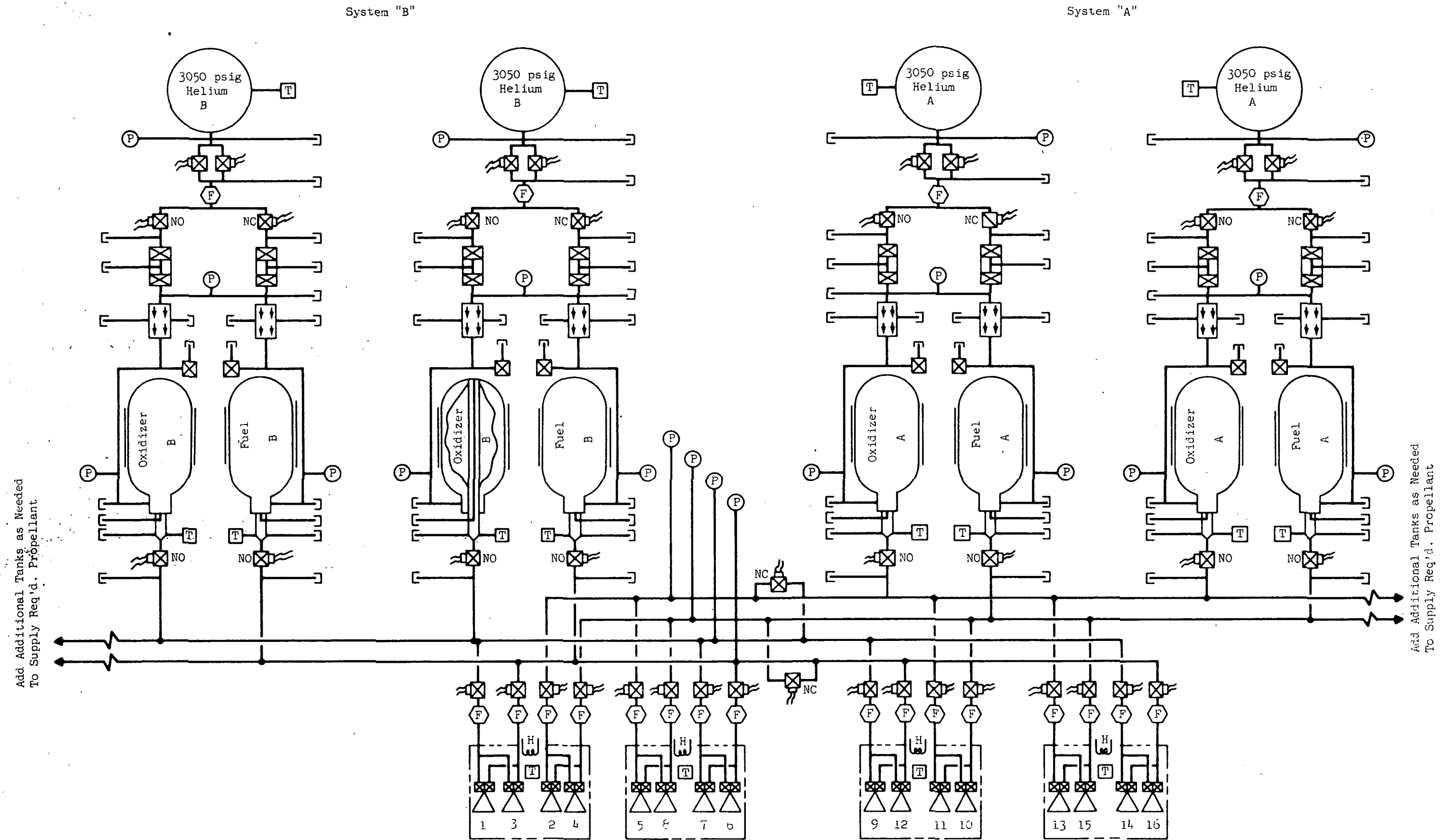


Fig. 3.5-1 Concept "A", Reaction Control Subsystem

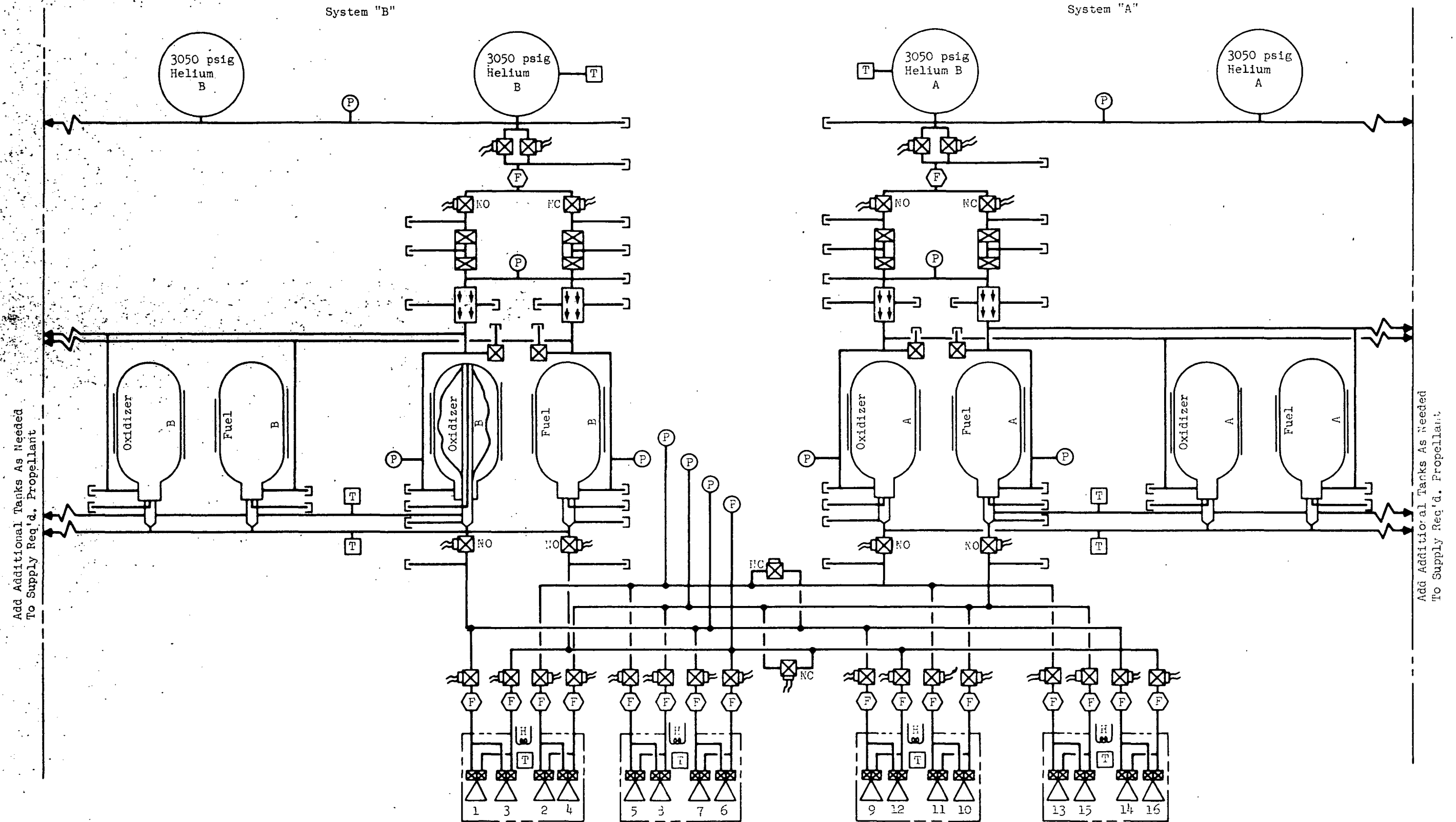


Fig. 3.5-2 Concept "B", Reaction Control Subsystem

3.6 CREW PROVISIONS & INTERIOR ARRANGEMENT

All crew provisions have been stored to allow maximum free space in the experiment and equipment bay areas. The term "experiment area" refers to the section containing displays and controls forward of LEM station Z + 27. The "equipment bay" refers to the area between stations Z + 27 and Z - 27. These are intended to serve as work areas only, and except for flights demanding complete "house-keeping", personal hygiene, recreation, eating and sleeping facilities will be provided in the CSM only. Existing primary structure of the ascent stage was maintained, except in those cases where an optical viewfinder system had to pass through the pressure wall. Wherever possible, existing secondary structure was used to allow attachment of similar LEM consoles. Where new or enlarged consoles were needed, secondary structure was added to existing pick-up points in the experiment area.

The specific arrangement of the crew provisioning depends upon the combination of experiments involved in each flight. The Biomed/Human performance experiments, docking, star tracking and artificial "g" experiments requiring attitude control, and those experiments involving an optical viewfinder system, dictate console arrangements, displays, and controls. The combinations of experiments, however, tended to produce four basic groupings of interior arrangements.

Four locations for control and display consoles in the experiment area were available--front, right side, left side and rear facing left side. Biomed/Human performance experiments were carried on all flights; the front console was used except when flight controls were required. In the latter cases, the Biomed controls were installed on the right side console, the full left side being unavailable due to the flight controllers. In cases where an optical viewfinder was carried in addition to flight controls, the rear-facing left side was utilized for Biomed, while the right side was used for the viewfinder console. The table below shows the combinations of console positions.

	<u>Front Console</u>	<u>Right Side Console</u>	<u>Left Side Console</u>	<u>Rear Facing Left Side</u>
Basic I	Biomed/Human performance basic lab and other experiments.	Other experiments or storage. Circuit breakers.	Work Top, storage, or other experiments. Circuit breakers.	Nothing
Basic II	Biomed/Human performance basic lab and other experiments.	Optical viewfinder experiments. Circuit breakers.	Other experiments or work top, Mod LEM controls. Circuit breakers.	Nothing
Basic III	Modified LEM flt controls and other experiments.	Biomed/Human performance & other experiments. Circuit breakers.	Small vertical panel for other experiments & fold away work top. Mod LEM. Circuit breakers.	Nothing
Basic IV	Modified LEM flt controls and other experiments.	Optical viewfinder. Circuit breakers.	Small vertical panel for Biomed/Human performance & fold away work top. Circuit breakers.	Biomed/Human performance.

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Each of these Basic Console Arrangements was carried on approximately four flights. Except for the control and display consoles, considerable commonality has been achieved in the location of other equipment. A summary of the location of these equipments follows:

ECS - The existing LEM ECS package remains on the right side in the Equipment Bay.

Lithium Hydroxide - Fourteen Lithium Hydroxide canisters for the CSM, in addition to those stored in the CSM, are stacked on Bulkhead Z-27, the remainder are stored outside, depending on mission life.

Food - Besides that stored in the CSM, additional food is stored below the floor in the experiment area. For flights beyond 14 days, the excess is stored in cavities around the ECS Package.

Biomed - Biomed/Human Performance experimental packages are stored in two areas - one in a pressurized pull-up can in the equipment bay, in place of the Ascent Engine. The other is in a cabinet below the right side console in the experiment area.

Circuit Breakers - Locations have been maintained left and right side in the experiment area, as in the LEM.

Lighting - The LEM spots and flood lights are retained in the same area.

Recorders - Recorders, film and tape storage, and experiment computer, as required, are stored on the left side in the experiment area.

Experiment Packages - The left side and bulkhead Z-27 in the equipment bay are utilized, as needed, for storage of experiment packages requiring pressurization.

Seats - Two adjustable, stowable swivel seats with restraint belts are provided in the experiment area, allowing for sit/stand operation of the controls. Restraints for the standing mode in zero-g will be developed in conjunction with the LEM program.

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3.7 CONTROLS & DISPLAYS

Whenever possible, the LEM Lab employs existing controls and displays in the same locations as in the Lunar Excursion Module. Table 3.7-1 lists the LEM controls and displays retained in the basic laboratory. Portions of the LEM flight control system are included in those flights which require flight controls. These controls are also located in the laboratory in the same positions as in the LEM, and are listed in Table 3.7-2.

Several basic arrangements incorporate displays and controls necessary to conduct experiments within the laboratory. All flights employ displays and controls for the performance of Biomedical and Behavioral experiments. The Biomedical portion consists of:

- Experiment plug-in connectors
- Two banks of lighted push-buttons to initiate recording of Biomedical Parameters
- Digital readout of
 - Heart Rate
 - Cardiac Output
 - Blood Pressure - Systolic & Diastolic
 - Body Temperature
 - Respiration Rate
 - Respiration Depth
 - Ear Oximeter output

In addition, an 8-inch cathode-ray tube is provided for detailed analysis of parameters. This tube serves a dual purpose. It is capable of displaying parameters required during "tracking" tests in the Behavioral portion of the experiments. It is also used for analysis of the Biomedical experiment parameters. The Behavioral experiments controls and displays are contained in the Human Performance Test Panel. This panel has provisions for:

- Numeric Display Monitor - digital readout of numerics for assessing addition and subtraction capability.
- Numeric Selector - push-button panel for input of numeric answers to addition and subtraction problems.
- Meter integrating monitor - vertical and circular meter displays and associated controls for reading adjustment and acknowledgement of acceptable reading.
- Multi-color light - Pushbutton Display - to assess response to a lighted display.

- Tape Recorder Controls - For reading verbal comments of the experiments and audiometer testing.
- Orthorater - for conducting visual experiments.

Three basic console layouts are used for the incorporation of the Biomedical and Behavioral experiments. Section 3.6 lists the positions of these experiments for the various interior layouts, and shows the relationship of display and control placement for the basic flight configurations.

Table 3.7-1

LEM Basic Controls & Displays as Used in Laboratory Concept

No.	Function	Display Type	Control Type
1.	Lighting Controls Anunciator/Numeric Lights Flood Lights Integral Lights Dome Lights Recognition Lights Side Console Lights		Rotary Rotary Rotary Rotary Toggle Switch Toggle Switch
2.	Caution & Warning	Indicator Lights	
3.	Alarm	Indicator Light	
4.	Test - Caution & Warning & Alarm		Illuminated Pushbutton
5.	Environmental Control System Suit & Cabin Temperature Suit & Cabin Pressure CO ₂ Partial Pressure Glycol Temperature & Pressure O ₂ Pressure & Quantity LiOH H ₂ O Separation Suit Fan Control Suit Fan Operating Glycol Pump Control	Panel Meter Panel Meter Panel Meter Panel Meter Panel Meter Indicator Light Indicator Light Indicator Light	Toggle switch Rotary step control

Table 3.7.-1 (con't)

No.	Function	Display Type	Control Type
6.	Audio Controls VHF ICS PLSS Emergency Voice Vox Vox Sensor S-Band S-Band		Toggle switch Toggle switch Toggle switch Toggle switch Toggle switch Rotary Toggle switch Rotary
7.	Cryogenic Storage Quantity Pressure Auto-Manual Open-close	Panel Meter Panel Meter	Toggle switch Toggle switch
8.	Power Generation & Distribution (Battery) Power & Bus Voltage Amperage Power Monitor Bus Monitor Battery Control	Panel Meter Panel Meter	Rotary Step Switch Rotary Step Switch Toggle switches
9.	Clock		
10.	Circuit Breakers		Lighted push-switch

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3.7-4

Table 3.7-2

LEM Flight Controls & Displays as Used in Laboratory Concept

No.	Function	Display Type	Control Type
1.	Pitch Rate	Panel Meter	
2.	Roll Rate	Panel Meter	
3.	Yaw Rate	Panel Meter	
4.	Attitude Controller		Formation Stick
5.	Translational Controller (where required)		T-Bar
6.	Reaction Control System		
	Temperature	Panel Meter	
	Pressure	Panel Meter	
	Propellant Quantity-Oxidizer & Fuel	Digital Indicator	
	Regulator Control		Toggle Switches
	Regulator position	Flag Indicators	
	Shut-off valve control		Toggle Switches
	Shut-off Valve position	Flag Indicators	
	Thruster pair control		Toggle Switches
	Thruster pair position	Flag Indicator	
	Temperature/Pressure Monitor		Rotary Step Switches
	Quantity Monitor		Rotary Step Switch
	Thrust Chamber Assembly Leak Test		Toggle Switches
7.	Stabilization & Control		Toggle Switch
	Deadband Control		Toggle Switch
	Rate Gyro Control		Toggle Switch
	Roll Mode Select		Toggle Switch
	Pitch Mode Select		Toggle Switch

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3.8 SUBSYSTEMS UTILIZATION SUMMARY

A summary of major subsystems carried on each of the NASA and DOD flights studied is shown in Figure 3.8-1.

The chart shows those flights in which a major portion of existing LEM subsystems are retained in the flight configuration. The basic LEM ascent stage structure and environmental control systems are used on all lab flights. The descent stage structure, stabilization and control system, reaction control system and instrumentation systems are used on many of the flights. Somewhat lesser utilization is seen for the Electrical Power, Navigation, Guidance, Communications and Propulsion Subsystems. The overall utilization of existing LEM subsystems is high.

The chart also shows those flights that require additional subsystem hardware that is not presently part of the basic LEM vehicle. These added subsystems make maximum use of existing Apollo spacecraft components.

TABLE 3.8-1

SUBSYSTEMS UTILIZATION SUMMARY

NASA FLIGHT NO. -	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16						
	209	211	507	509	215	513	218	219	221	516	518	521	523	229	230	-	DOD 1	DOD 2	DOD 3	DOD 4	DOD 5	DOD 6
LEM H'DW'RE UTILIZED																						
• Structure - Asc. Stage	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X	X	X	X
• - Desc. Stage	X	(X)	X		X	X	X		X	X	X		X		X	X	X	X	X			
• Electrical Power	X															X	X					
• Stabilization & Control	X			X	X		X		X	X	X		X			X						
• Reaction Control	X			X	X		X	X	X	X	X		X	X		X						
• Guidance and Nav.																X						
• Environ. Control	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X	X	X	X
• Communications				X	X		X		X				X			X						
• Instrumentation	X	X	X	X	X	X	X		X	X	X	X	X			X				X	X	X
• Crew Systems	X			X	X		X		X	X	X		X			X						
• Propulsion																X						
PRINCIPAL ADD'L H'DW'RE																						
• Struct. - Low Desc. Stage			(X)	X				X				X		X						X	X	X
• - Airlocks																				X	X	X
• Pwr. Supply - FCA, Rad.		X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X	X	X
• ECS - Rad., Water Man.		X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X	X	X
• SCS - Attitude Sensor Syst.		X	X		X	X				X	X	X	X				X	X	X	X	X	X
• Instr - Data M'ngm't Syst.	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X					X	X	X

(X) Alternate Designs

4 SUPPORTING STUDIES

4.1 THERMAL CONTROL - LEM LAB

4.1.1 Summary

The LEM is reasonably independent of external thermal environment. Environmental extremes in LEM Lab missions are less severe than the present LEM lunar missions. The basic LEM structure is insulated from an outer thermal radiation shield, to minimize the influence of outside environment. Equipment is generally hard-mounted to structure and nominally runs at structural temperature. The LEM Lab power dissipating equipment is cooled by a glycol system which transfers the heat to a space radiator and water boiler. A variety of active and passive thermal control methods are also used for temperature control of items such as cryogenic tanks, fuel cells, IR detectors, and optical experiments.

Development of a proper by-pass and shut-off system for the radiators allows for some very attractive simplifications in the design of the many different LEM Lab vehicles. The concept for such a controlled radiator has been partially evaluated and appears feasible. Implementation of a slightly over-designed controllable radiator system will, therefore, allow the LEM Lab to be insensitive to orbit inclination, altitude, vehicle orientation, configuration, structural details, and large power profile variations.

The thermal development test program depends on the selected thermal design. The required test program for each LEM Lab mission would be intensive for an all passive thermal system; an active, controllable thermal system will require a less exhaustive program.

The following sections review the general concepts used in the thermal design, and then the more important design areas are reviewed in slightly more detail.

4.1.2 General Concept

Main structure is insulated from the external thermal radiation skins, and the equipment is hard-mounted to the structure (except for unusual items). Excess heat dissipation in equipment is transferred to a glycol cooling system. The basic LEM has a net heat loss of about 300 watts (1000 BTU/Hr), by passive means, from the structure to space. As a function of the specific design, the passive structural loss for the various LEM Labs can vary from a minor amount to twice as large as the basic LEM. Passive heat losses are not relied upon (except for Flight 209) and, therefore, these results tend to be conservative.

Radiators are sized for average load, and water boiling is used for the peak power loads. In most cases, all the water generated by the fuel cell is not used. The final design will use this excess water for back-up thermal control, and to help smooth out orbital variations and additional peaks in the power profile. The radiator size depends on the mission, but will be between the following extremes: equipment radiator - 28.9 to 97.5 ft² per KW; fuel cell radiator - 4.7 to 27.8 ft² per fuel cell.

The radiators will be designed in standard sizes, and multiples of this standard will be used. Most missions will thus be partially oversized. The basic radiator design is such that its capacity can be decreased (even under the environmental extremes) to a small fraction of its design capacity. The by-pass control system handles the coolant freezing problem in a selective manner.

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External surfaces of the fuel cells structure are passively cooled by radiation to an external LEM Alzak skin.

Excess cryogenic boil-off will be eliminated by passive thermal methods, or, in exceptional cases, the power profile will be modified.

Certain experiments will be thermally controlled separately. IR detectors will be cooled by stored cryogenics or active refrigeration units. Optical experiments are to be controlled by a combination of strip heaters, insulation, and passive design concepts. External antennas, structures, balloons, etc. will be controlled by passive concepts.

Passive thermal control of the entire LEM Lab heat load can be approached in some of the Lab missions. This requires stringent restraints on equipment location and structural design. Such a design requires full scale thermal vacuum development testing of most of the Lab configurations. The increased reliability of a completely passive system sometimes justifies its use. For this study a slightly oversized active system is utilized.

4.1.3 Cryogenic Tanks

Several ground rules were established to facilitate selection of the types of cryogenic storage tanks (Block II or Block III) to be used, and the evaluation of the amounts of H_2 and O_2 to be loaded. These ground rules are:

- Obtain minimum wet storage system weight.
- Use average power profiles and mass flow rates of reactants, based on 150°F environmental temperature to evaluate waste boil-off.
- Minimize waste, where possible, by off-loading or overloading storage tanks.
- Minimize waste, where possible, by allowing off-loaded H_2 tanks to obtain operating pressure by using the natural heat leak only.
- Assume waste O_2 boil-off can always be vented through the cabin repressurization system.
- Assume that if H_2 waste boil-off cannot be prevented by control of the environmental storage temperature, the power profile will be modified to accommodate the boil-off. Venting of H_2 gas will not be permitted.

A summary of the pertinent data obtained from the cryogenic tank study is presented in Table 4.1-1. Both the ECS and fuel cell requirements are included in the usable pounds of oxygen. The last four columns present data concerning feasibility of eliminating waste reactants. These data are (1) the no-waste environmental temperature, i.e., the environmental temperature at which the mass flow rate of hydrogen leaving the tank due to the natural heat leak does not exceed that required by the power profile, (2) the minimum obtainable environmental temperature, (3) a conclusion from columns (1) and (2) regarding the necessity for changing power profile, (4) an indication as to the feasibility of hard mounting the tanks to the LEM structure (no waste for an environment of 590°R or greater).

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Performance characteristics of the Apollo Block II and III (Apollo-X) hydrogen and oxygen cryogenic storage tanks for a 150°F environment are shown in Figure 4.1-1 a, b, c, d. These data were obtained from the information presented in references SID 64-1860-9 and addendum to NAS 9-3140. Figure 4.1-2 presents the time required for the natural heat leak to increase the tank pressure from the loading pressure to the operating pressure for Block II and III hydrogen tanks. These times are based on the specification values for the standby heat leak.

Tanks which produced no waste, based on the thermodynamic data of Figure 4.1-1 (environmental temperature equals 150°F), may be hard mounted to the structure. (LEM structure will nominally remain between 30°F and 130°F.) It must be noted that if all cryogenic storage tanks are hard-mounted, some flights will require additional masses of oxygen to burn all the H_2 boiled-off. For example, DOD-3C would require an additional 84 lbs of O_2 . All weight factors considered, it is almost always lighter to insulate and eliminate the waste than to burn it.

4.1.3.1 Waste Control By Selective Loading

Figure 4.1-3 shows two typical analyses of the cryogenic tankage. Flight DOD #1 Conf. C is presented as an example of selective loading in a two-tank system to eliminate waste. On this flight, 450 lbs of oxygen are required. If both tanks are evenly loaded, the initial density is 48.6 lb/ft³. Starting from the point on the flow rate curve which corresponds to a density of 48.6 lb/ft³, and moving to the right, the waste is shown to equal the area bounded by the flow rate curve and power profile requirement. However, if one tank is full (density equals 68.6 lb/ft³) and the other is off loaded (density = 28.8 lb/ft³), the flow rate vs time - due to the natural heat leak - is found by adding the mass flow rates as presented in Figure 4.1-1, using the two densities as the initial starting points. The result is the dashed line in Figure 4.1-3. This curve is always below the power profile requirement; hence, there is no waste.

Flight 221 Conf. D is a case in which both selective loading and self-pressurization are used to minimize waste. The solid line in Figure 4.1-3 presents the characteristics of selectively loaded tanks, each at operating pressure at the time of lift-off. The dotted line presents the flow rate history (starting from 14 psia) when the off-loaded tanks is allowed to be self-pressurized. The cross-hatched area indicates the reduction in the waste.

4.1.3.2 Waste Control By Environmental Temperature Selection

Flights which have excess cryogenic boil off when subject to 150°F environment can have this waste eliminated in a lower temperature environment. Figure 4.1-4 is used to determine this required lower temperature. The ratio of required boil off rate to boil off at 150°F is proportional to the ratio of desired to design heat flow rate in Figure 4.1-4.

The obtainable environmental temperature in a well insulated storage bay having an uninsulated external Alzak skin is presented in Figure 4.1-5. The compartment was assumed to be insulated with 25 layers of NRC-2 superinsulation with a wrapping factor of two. For the largest ratio of insulated compartment area to skin area which might exist, a maximum of 5 btu/hr per sq ft of skin area could be transferred by LEM at 590°R to the compartment at 0°R. Development testing of similar systems at Grumman has indicated that good design practice results in conduction heat leaks which would double this value. Thus, 10 btu/hr/ft² represents a maximum LEM heat

load which could be rejected by the skin. Temperatures for the cryogenic tankage and skin are presented in Figure 4.1-5 as a function of the absorbed external radiation. The range of absorbed radiation for each type of flight is indicated, and the resulting obtainable environmental tank temperatures are listed in Table 4.1-1. In most cases the maximum environmental temperature is less than that required to eliminate waste, if the tank location restrictions are acceptable.

Insulation will eliminate waste, if the range of obtainable environmental temperatures listed in Table 4.1-1 for an insulated storage bay (column 2) is less than the no waste temperature (column 1). On some flights the no waste environmental temperatures is within the range of obtainable temperatures. For these flights, waste quantity depends upon the location of the cryogenic storage bay and vehicle orientation. For such a case it is questionable if there is a need to modify the power profile to prevent venting of waste. On Flight 521-D, 218-D, the no waste temperature falls below the lower limit of the obtainable temperature. Modification of the power profile is the most logical approach to handle the waste and prevent venting.

4.1.4 Fuel Cells

If not converted to electricity, the excess heat generated by the fuel cell must be removed from the unit, so as to prevent excessive temperatures. Heat and water vapor generated at the electrodes are carried by a closed internal hydrogen cooling loop to a hydrogen, glycol heat exchanger. Heat picked up by the glycol is dissipated in a radiator and a water boiler. The radiator is sized to handle the average fuel cell power for the given flight; the water boiler is used as a supplement for peak loads. Some heat leaks through the fuel cell insulation and is radiated from the outside surface of the cell to the environment. Both active and passive modes of heat dissipation are considered.

Radiator sizes and locations, as water boiler requirements are listed in Table 4.1-3 for each flight and configuration. All water boiler requirements are supplied by excess water generated in the fuel cell. Fuel cells will be located in bays having an external space-exposed skin. The cells will also be thermally insulated from LEM structure and all other equipment. Approximately eight sq ft of Alzak skin area are required for each fuel cell, to insure that the external surfaces of the cell do not exceed the 190°F temperature imposed by fuel cell structural limitations.

4.1.4.1 Active Cooling Requirements

The required radiator capacity is determined by using the average fuel cell power from Table 4.1-3 and the glycol cooling requirements per fuel cell as shown in Figure 4.1-6.

The fuel cell radiator inlet and outlet temperature are fixed by P & W performance data on the CM fuel cell. The initial radiator sizing is, therefore, based on 210°F and 119°F for the design point inlet and outlet radiator temperatures. Figure 4.1-9 gives capacity of the fuel cell radiator per unit area as a function of the radiation absorbed from external environment.

Radiator size is obtained for the maximum externally-absorbed radiation of the given mission and radiator location. The resulting radiator size, location, and water boiler requirements are listed in Table 4.1-3 based on the orbital parameters on vehicle orientation in Table 4.1-2. All the variables of external environment, and the extremes in fuel cell power, are considered in the following table which lists the possible extremes of radiator size per active fuel cell. All sizes in Table 4.1-3 are within these limits.

Fuel Cell Radiator Size
(Ft. ²/Fuel Cell)

	Extreme Environment	
	Max.	Min.
Max. Power (1420 w)	27.8	14.6
Min. Power (560 w)	9.07	4.78

The difference between the maximum fuel cell power (1420 w) and average power cooling requirement is used to determine the water boiler needs. The remainder of the peak power is supplied by other means, such as a battery, and its excess heat is handled by the equipment cooling system.

4.1.4.2 Passive Cooling Requirements were established on the basis that the external surface of the fuel cell is not to exceed 190°F while dissipating 550 BTU/hr. This temperature is imposed by fuel cell structural considerations. Figure 4.1-7 presents the temperature of a unit area of surface dissipating heat to a unit area of Alzak skin as a function of the amount of heat being dissipated, and the view factor for the minimum and maximum amounts of external heat absorbed. The required area of operation is shown on the figure. To insure proper operation, a high value of emittance for the outer surface of the fuel cell and the inner surface of the Alzak skin is required. Consideration was also given to the problem of dissipating the necessary amount of heat from the rear surface of the cell without incurring large temperature gradients on the cell. To evaluate steady-state temperatures on the front and rear of the fuel cells, the following configuration was considered. Fuel cells were approximated by right circular cylinders in a rectangular enclosure well insulated on five sides, the sixth side being an Alzak skin with a maximum external heat input of 82.6 BTU/ft²hr. It was determined that the minimum practical spacing between individual fuel cells and walls is about six inches, and results in a temperature difference of 11°F between the front and rear of a cell. When three cells are operating at maximum power, rejecting 550 BTU/hr. passively, a maximum temperature of 202°F was obtained on the rear of the center fuel cell. On the basis of this spacing eight sq ft of Alzak skin will be required for each fuel cell. To insulate the LEM structure and other equipment from the fuel cell heat loss, and to provide a highly reflective surface inside the bay, 25 layers of NRC-2 superinsulation will be used on all except the Alzak skin surfaces of the fuel cell bay.

4.1.5 Active Thermal Control Of Equipment

The heat generated by the electrical equipment will be cooled by a glycol system which finally dissipates the heat to a space radiator and supplementary water boiler. For the purpose of initial estimates, the radiator is sized for the average power supplied to the LEM, and water boiler requirements are based on the peaks above the average power. The design powers, and resulting radiator size, location and water boiler requirements are listed in Table 4.1-3.

Using radiator inlet and outlet temperature of 100° and 40°F , respectively, the radiator capacity per unit area is determined from Figure 4.1-9 as a function of absorbed external radiations. Considering the extremes of vehicle orientation, orbit parameters, (Table 4.1-2) and radiator location, the possible extremes on radiator size run from 28.9 to 97.5 ft^2/kw of waste heat. Since the larger radiators consist of sections located on a number of sides of the vehicle, the average vehicle radiator requirement will more closely approximate 40 ft^2/kw . The extremes on radiator size for specific orbits can be determined from Figure 4.1-8.

The present LEM has a net structural heat loss in the orbital space environment. This heat loss will finally show up as a decrease in the cooling requirements that have to be handled by the radiators and water boilers. As a round number, this heat loss is in the order of 300 watts. For the various missions proposed for the LEM Lab, this heat loss could easily be doubled or cut to a small fraction. This heat loss variation is a function of the external configuration and the number and type of support structure that pierce the vehicle skin to tie down external components. Since there will probably always be a heat loss, the initial radiator and water boiler estimate are conservative. Considering the weight of the radiators, and the tests required to determine the actual heat leakage for each mission, it has tentatively been decided to overdesign the radiators, and the built in radiator control system will adjust for the difference.

4.1.6 Radiator System Flexibility

The following section discusses how the radiators are sized for the various missions, based on the design average power, and how the radiator responds to decreases in the vehicle power dissipation and externally absorbed thermal radiation. Finally, a discussion is included on the method proposed to solve the radiator freezing problem. This flexibility in radiator performance, and continued radiator operation with partial freezing allows the LEM Lab to be reasonably insensitive to the changes involved in each mission.

4.1.6.1 Radiator Capacity

The heat dissipating capacity of a radiator located on a vehicle in Earth orbit is influenced by the following factors: (1) coolant inlet and outlet temperature, (2) radiant energy emitted by the sun and Earth and absorbed by the radiator, (3) radiator geometry and surface properties. The inlet and outlet temperatures of the water-glycol mixture circulated through the radiator depend on the particular system being cooled. The operating temperatures for the fuel cell radiator are determined by fuel cell performance requirements, and the operating temperatures for the ECS radiator are determined by allowable equipment temperature on the hot side and cabin-air dehumidification requirements on the cold side.

The absorbed environmental radiation is a function of the radiator surface properties, and vehicle orientation and orbit parameters in a particular mission. For the particular thermal properties used and missions considered, absorbed radiation varies between 0 and 100 BTU/hr/ft² as indicated below.

External Radiation Absorbed By Radiators
(BTU/hr/ft²)

Orbital Parameters	Hot Side	Sphere	Cold Side
200 n mi i = 28.5° 0° solar vector Sun Oriented	71.49	36.79	21.42
Earth Oriented	37.3	36.79	21.42
200 n mi i = 81.5° 90° to solar vector Sun Oriented	99	41.83	18.9
Earth Oriented	99	41.83	0
Synchronous i = 0 0° solar vector Sun Oriented	75.6	19.02	0
Earth Oriented	24.0	19.02	0

The above table is based on 0.18 and 0.9 for the absorptance and emittance of a radiator coating consisting of zine oxide in potasium silicate (Zn O in K₂ Si O₄). The external radiation sources used (direct earth and solar, and reflected solar) are the orbital average values.

Radiator fin effectiveness relates capacity to geometrical considerations such as radiator thickness and tube spacing. Using 0.020" thick aluminum panels and spacing the tubes on 4 in. centers, results in a radiator fin effectiveness of 0.96. Figure 4.1-9 shows the variation in radiator capacity with absorbed radiation for both the fuel cell radiator and several ECS system radiators.

The design condition used in selecting the ECS system radiators for the various missions in 100°F inlet - 40°F outlet temperature and the comparable fuel cell radiator values are 210°F and 119°F.

4.1.6.2 Performance of Radiator in Off Design Conditions

Having selected a radiator for a particular design condition it is important to know its performance during off-design operation. The fuel cell radiator is less dependent on environmental conditions because of its higher operating temperatures. Therefore, discussion of performance will be limited to the ECS System Radiator. Transient analysis was performed to determine variation in equipment temperature resulting from varying amounts of absorbed radiation during an orbit. Results for a rather severe orbit showed maximum variation of 10°F from the design value. Therefore, the orbital transients will mainly affect the radiators, but not the equipment.

Figure 4.1-10 shows the steady state performance characteristics of the ECS System Radiator for different heat rejection loads and radiation environments. The performance characteristic is calculated using a closed solution to the one-dimensional radiator equation for a constant coolant flow rate and the published numerical integration of the fin effectiveness equation. Radiator operation at low capacity and/or in cold environments results in low operating temperatures. It is assumed that the glycol-water mixture flow below -30°F is not desired. Operation at outlet temperatures lower than -30°F is, therefore, precluded. Radiator operating conditions resulting in the -30°F outlet temperature are shown by the heavy line in Figure 4.1-10. The permissible operating range of the radiator is shown by the solid lines to the right of the -30° outlet line. To the left of the -30°F outlet temperature line, the performance lines are shown dashed, indicating the region in which undesirable radiator temperatures exists.

4.1.6.3 Radiator Capacity Control by Flow Regulation

The equipment radiator cannot operate over the entire anticipated range of environment and capacity, therefore, some method of capacity control is required. One method of capacity control consists of changing radiator fin effectiveness by changing fin length. This is accomplished by diverting the flow of coolant to one large tube located at the edge of the radiator and allowing the coolant to freeze in the other tubes. Figure 4.1-11 shows the fin length required to maintain the outlet temperature of the coolant flowing in the bypass tube at -30°F . Selection of a 36 in wide radiator permits operation in this manner to zero absorbed radiation with 30 percent capacity.

The predicted performance of a 36 in. wide radiator for various environments and with bypass operation is presented in Figure 4.1-12. The two cases considered are 100 percent capacity and 30 percent capacity. In this figure the solid lines show the variation in radiator inlet temperature with the bypass fully closed. As the absorbed radiation decreases, the radiator outlet temperature get colder. When outlet temperature reaches -30°F , the bypass valve opens fully, diverting the coolant flow through the bypass tube and changing the radiator effectiveness. This results in the step change in temperature. Radiator performance for bypass operation is shown by the dashed lines in the figure. Such a step change is not acceptable, and is eliminated by a modulated control which will perform in the shaded area of Figure 4.1-12.

As an example, the performance of the ECS system radiator will be discussed for a particular variation of capacity and environment. The assumed process is shown superimposed on the radiator performance curve in Figure 4.1-10. The radiator is initially operating at the design point of 100 percent capacity with 70 BTU/hr ft^2 absorbed radiation. This is labeled 1 in the figure. A decrease in capacity from 100 percent to 30 percent of design capacity at a constant value of absorbed radiation results in a decrease in radiator temperature along the vertical line from 1 to 2. The load is then held constant at 30 percent of design capacity and the absorbed radiation decreases until the outlet temperature is -30°F . This process is shown by the line between 2 and 3. When the outlet temperature is -30°F , the bypass valve diverts the entire coolant flow through the bypass tube, thereby reducing the fin effectiveness and resulting in the jump from 3 to 4. A continued decrease in absorbed radiation with constant 30 percent capacity and full by-pass operation is shown by the line from 4 to 5.

Using the same logic and control for 100 percent capacity in one radiator results in undesirable temperature response. The actual flow control (discussed in 4.1.5.4) is a modulated one. It has been assumed that the by pass valve starts to open at $+30^{\circ}\text{F}$ radiator outlet temperature and is full open for 100 percent bypass at -30°F with a linear temperature relationship in between. The exact calculations for this combined condition have not been performed at this time, but bracketing type calculations have been made. In figure 4.1-12 the shaded area represents roughly the temperature of the radiator inlet for the modulated flow conditions. For the 100 percent capacity curve used as the example, the mix outlet temperature is 60°F cooler than the inlet. The outlet temperature of all the tubes will be different, the by pass outlet the highest temperature, and each adjacent tube outlet colder and colder until finally the tube is frozen shut. The resulting performance will be a smooth adjustment to load and environment changes.

4.1.5.4 Description of System Operation

Figure 4.1-13 describes a system of several radiators connected in parallel in a single glycol loop. It is assumed that the radiators occupy various faces of a spacecraft and hence are oriented in different ways with respect to solar and/or planetary radiation. Each radiator, therefore, receives, glycol from the common supply manifold at the same temperature but returns glycol to the common return manifold at a temperature dependent on the cooling capacity of each radiator. Flow rate through each radiator tends to be the same, but is affected by temperature, and hence viscosity, of the fluid within each radiator. The return temperature from the entire radiator array is therefore the resultant mix of the various fluid temperatures and flows returned.

Practical design considerations limit the acceptable range of glycol temperatures returned by the radiator system. The range from $+60^{\circ}\text{F}$ to -30°F is used for the present example. The upper limit represents maximum load combined with solar and planetary absorption and constitutes the radiator system design point. The lower limit represents the minimum acceptable glycol temperature out of radiators from a flow stability standpoint. Moreover, since the lower temperature limit is for reasons of flow stability, it is applicable to any single radiator panel and not simply to the resultant mix temperature of all returns combined. This lower limit of acceptable return temperature is not the lowest temperature which could be returned by an uncontrolled radiator with minimum load and absorption. A control means is therefore required in a practical system, applicable to each individual radiator panel.

Of the several possible control means which have at various times been proposed, one is chosen for presentation here which is relatively simple and reliable, using equipment designs similar to those already existing in the LEM vehicle. The basic radiator contains a number of tubes of uniform size and spacing, connected in parallel from a common inlet header to a common outlet header. A separate single tube of much larger size is situated near one edge of the tube sheet, to be used as a by-pass. The by-pass flow mixes with the main radiator flow in a by-pass control valve situated at the outlet. The ratio between main and by-pass flows is modulated by the by-pass valve in response to the temperature of the mixture. A proportional schedule of flow ratio versus mixed temperature is built into the valve, the range of proportional control lying within the range of acceptable return temperatures.

The operation of the control is as follows: As load, and/or absorbed radiation decreases, radiator return temperature similarly decreases. When the return temperature has decreased sufficiently to come within the control range, the valve by-passes flow in progressively greater proportions as return temperature continues to decrease. The effect of by-passing flow is to increase return temperature through decreasing overall radiator effectiveness. The valve position at which the temperature sensor is satisfied by the resultant return temperature is the equilibrium valve position for the particular load and absorption condition existing. Dynamic stability of the control is enhanced by a wide temperature band of proportionality in the control. It appears that a band as wide as $+30^{\circ}\text{F}$ to -30°F may be acceptable from an overall system operational standpoint. Such a wide band width provides a large measure of proportionality, plus a large amount of available energy from the sensing element, making a simple, direct-acting control practicable with no force amplification.

The control valve depicted in Figure 4.1-13 utilizes a spool valve for modulating glycol flow entering at ports (A) (main radiator) and (B) (radiator by-pass) and leaving through a common port. The spool is positioned directly by the temperature-sensitive element, which is a wax-filled thermal actuator, providing a linear movement proportional to temperature. The mixed flow leaving the control section of the valve is routed over the thermal actuator, completing the feedback loop. The valve depicted is only one of many design approaches which will provide the required control function. It is chosen as the illustration of the principle because of its inherent simplicity and because it is identical in construction and operation to the LEM cabin temperature control valve. Questions of design suitability and reliability, when answered for the cabin control valve, will be largely answered for the suggested radiator control valve also.

The effect of the described by-pass controls, operating independently on each radiator panel, is that each panel will return glycol to the system at some temperature within the prescribed range. The resulting mix of all return flows will therefore be within the prescribed range also. As stated earlier, the lower limit of this range (-30°F) is established by flow stability considerations, connected with the slope of the viscosity-temperature curve for glycol/water mixtures and the proximity to the freezing point. This lower limit temperature, however, is much too low for practical use in the spacecraft for such purposes as cabin air cooling and dehumidification and is not required for other purposes such as equipment cooling. Owing to this, a regenerative heat exchanger, complete with by-pass control, is shown in the figure as an additional system feature. This heat exchanger will enable the portion of the loop on its radiator side to operate at a lower temperature than the portion of the loop on its cooling load side. The effectiveness of this heat exchanger is modulated from maximum to zero by a temperature-sensitive by-pass valve identical in design and construction to the radiator control valves described above, differing only in the temperature range setting.

The effect of the described regenerative heat exchanger and control valve is to narrow the range of glycol temperatures returned by the radiators to the spacecraft to the range ($+35^{\circ}\text{F}$ to $+60^{\circ}\text{F}$) under all combinations of load and external absorption.

4.1.7 Experimental Equipment

The large array of experimental equipment considered for the LEM Lab presents thermal requirements ranging from routine to difficult problems. Where-ever possible, the experiments are incorporated in the main cooling system of the LEM, but a number of special requirements are handled separately.

The special thermal control requirements of experiments are indicated by flight numbers in Table 4.1-4. The special thermal control items, are those which are outside the direct or indirect control of the glycol loop. Values listed in the Table are estimates of average electrical power required to maintain thermal control, and of weight of material and equipment chargeable to special thermal control. The weight penalty for electrical power generation is not included in these weight numbers. The thermal design concept considered for each class of experimental equipment from Table 4.1-4 is reviewed.

4.1.7.1 Infra-Red Detectors:

The required operating temperature of a given I-R detector is dependent on the junction material and the maximum wavelength it is desired to detect. Figure 4.1-14 is a generalized survey of a range of detector types in terms of detectivity versus wavelength. The cost of cryogenic cooling in terms of system weight increases with increasing heat leakage into detector from ambient and increases rapidly with decreasing temperature level required. Figure 4.1-15 illustrates the latter relationship for an arbitrarily-selected heat leak rate and mission duration. Closed-cycle mechanical systems are competitive with storage systems in certain temperature ranges. Closed-cycle systems may also have the advantage of portability, permitting a detector, together with its cooler, to be installed in or removed from an optical system during space flight.

The heat leak rate into a detector varies with its size, operating principles and method of construction. A single-junction detector, such as is illustrated schematically in Figure 4.1-16 may have a heat leak rate in the order of 1 watt, while an orthicon imager may have a heat leak in the order of 5 watts, including the I-R signal itself and the input through lead cables. Thus, for an extended mission, seemingly minor changes in the equipment selection and installation can have a significant effect on system weight penalty.

The source of cooling, whether a tank of liquified gas (open loop) or a mechanical refrigeration unit (closed loop) must be located in very close proximity to the detector in order to minimize heat leaks in interconnecting lines. Standards of acceptability for earth-based designs are not applicable to extended space missions owing to the high cost of cooling in terms of system weight. An example of a suggested installation of an I-R detector for open loop cooling is shown in Figure 4.1-17. A liquid hydrogen tank is provided especially for the purpose of cooling the detector. The detector is, in effect, "built in" to the tank wall, rejecting heat directly to the fluid in the tank itself. The hydrogen is maintained at a constant, sub-critical pressure, vented through a tank pressure regulator which serves as an expansion valve, followed by a built-in heat exchanger. This arrangement prevents loss of liquid in a zero-gravity environment, while maintaining the tank temperature constant throughout the use cycle.

4.1.7.2 Optical Equipment

Telescopes and large mapping cameras are basically designed to operate at ordinary room temperatures but require temperature control within very narrow limits for accurate maintenance of focal length and optical alignment. Typical allowable temperature variation around a designated design temperature is $\pm 10^\circ \text{F}$. These limits refer primarily to the optical system itself and to the structure connecting parts of the optical system to other parts and to the film plane.

Maintenance of such narrow temperature limits by passive means alone probably would impose severe constraints on flexibility of design. A more practicable approach is a semi-passive system in which a heat flow path exists which is variable in magnitude though not in direction. The proper application of insulation and coatings to the telescope or camera structure and mounting can cause a continuous heat loss to space, variable with sun and earth orientation but never resulting in a heat gain. The heat loss is then made up by many small electrical heaters distributed over the structure, each controlled by a small thermostat. The net result is control of structural temperature approximately within the control range of the thermostats.

In the case of telescopes, solar radiation is normally excluded from entering the open end of the tube by a sunshade and by restrictions on pointing angle. Planetary heat emitted and solar heat reflected from Earth may enter telescopes and cameras through the optical system and in most cases will be largely absorbed inside. Exposed surfaces must have sufficient area for dissipation of this heat to space by means of coatings of low absorptance to emittance (α/ϵ) ratio. An unfavorable combination of design factors can still be accommodated by selecting a higher design temperature, the penalty being a higher average heater power.

4.1.7.3 Passive External Equipment

Antennas are generally characterized by mounting location external to the spacecraft, relatively large surface area-to-weight ratio, low heat output and wide temperature tolerance. In general, the temperature tolerance range for antennas is approximately $\pm 250^\circ \text{F}$. Many coatings are available which will provide equilibrium temperatures within this range on a purely passive basis. Bare, polished metal surfaces are unsatisfactory because of high equilibrium temperature when exposed to the sun. The range of temperatures is reduced whenever heat absorbed on the side exposed to solar input can be conducted through the body and be emitted from the side in shadow. This occurs readily in the case of simple metal fabrications but less so for complex structures such as honeycomb constructions. In the case of these, heat must be transmitted across the cells by radiation, reducing the flux transmitted and increasing the spread of temperatures between the hot and cold sides. Temperature range limitation of honeycomb structures is therefore more dependent on properly controlled coatings than simple metal structures.

Inflatable Structures such as balloons are normally brought up to orbital altitude stored in a canister, which is then opened and the balloon deployed. Temperature limits of the deployed balloon are in the order of $\pm 250^\circ \text{F}$, which can be maintained passively by the coatings employed (example: Echo II). While in storage in the canister, temperatures above the freezing point of water are to be maintained, in order to eliminate any possibility of deployment failure being caused by ice. Allowable range of canister temperature is given as 35°F to 140°F . This range can be maintained passively by insulation, coatings and some thermal coupling to

spacecraft structure in the case of a partially-submerged canister installation. In the case of an externally-mounted canister, some electrical heat may be required.

4.1.8 Comparison of Alternate Equipment Thermal Control Methods

A number of methods can be used to carry away the heat dissipated by the electrical equipment. The proper system for a given vehicle depends on a variety of items: 1) extremes in electrical power dissipation and temperature requirements, 2) cooling system reliability, 3) testing cost and schedule, 4) flexibility to large changes in external heat load, vehicle configuration, and structure design.

It has been tentatively decided to emphasize the last two items. The main purpose being to be as adaptable as possible to a large array of types of experimental packages, orbital parameters and configuration changes. At the same time it is desirable to limit full scale thermal vacuum development test requirements for each mission. The primary cooling system has, therefore, been selected as an active glycol cooling system to carry heat from the equipment and dissipate this heat in a space radiator and supplementary water boiler.

In selection of a thermal control system the first item to be considered is the power density (amount of heat dissipated per unit area based on the area of one of the sides of the equipment). Figure 4.1-18 shows the allowable maximum power dissipation per unit area as a function of the type of thermal control system. The orbit selected as an example, is a synchronous orbit with random vehicle orientation, but fixed in space, with respect to the sun, for long periods. The net result being that a given surface can be both normal to the solar vector, or in complete shadow during a given mission.

For low power densities (equal or less than $52.5 \text{ BTU/ft}^2 = 15.4 \text{ watts/ft}^2$), System 1 can be used. This is a passive system, with the equipment radiating from one surface to the local alzak skin, which in turn radiates to space. Having determined the maximum power for this system, then the minimum power cannot be less than about 65 percent of this level ($Q_{\min} = 0.63 Q_{\max} = .63 \times 52.5 = 33 \text{ BTU/hr-ft}^2$). Use of moving vanes between the equipment and skin will allow a much lower minimum power (in the order of 10 percent of $Q_{\max}$).

System 2 allows the equipment also to radiate out the back surface into the vehicle. This allows the equipment to dissipate more heat on the hot side, and less heat on the cold side. System 3 is a further expansion on System 2. In this case an active system is used to circulate a coolant through the equipment to smooth out the temperature extremes between hot and cold sides. The ultimate method of heat rejection is still radiation from one surface of the equipment to the local skin area and then to space. System 3 can allow about 81.5 btu/hr-ft^2 (26 watts/ft^2) maximum power and 28 percent of this for minimum power.

For equipment designed with higher power densities in the order of 100 watts/ft^2 System 4 can be used. This System uses two coolant tubes, one on each side of the equipment. For much higher power dissipation, the entire surface on one side of the equipment must consist of cooling passages. (System 5).

Combination of Systems 4 and 5 is tentatively proposed for the LEM Lab missions. Such Systems can handle any lower level of power density, but the lower-numbered thermal control systems do not have the flexibility to handle the higher levels.

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4.1-14

Figure 4.1-19 represents the operating characteristics of Systems 1, 2, and 3. The allowable power densities of Figure 4.1-18 were based on the noted maximum and minimum temperature limits of $+130^{\circ}\text{F}$ and 0°F , respectively. For any other power level and the extremes of orientations noted, the resulting equipment temperatures can be determined.

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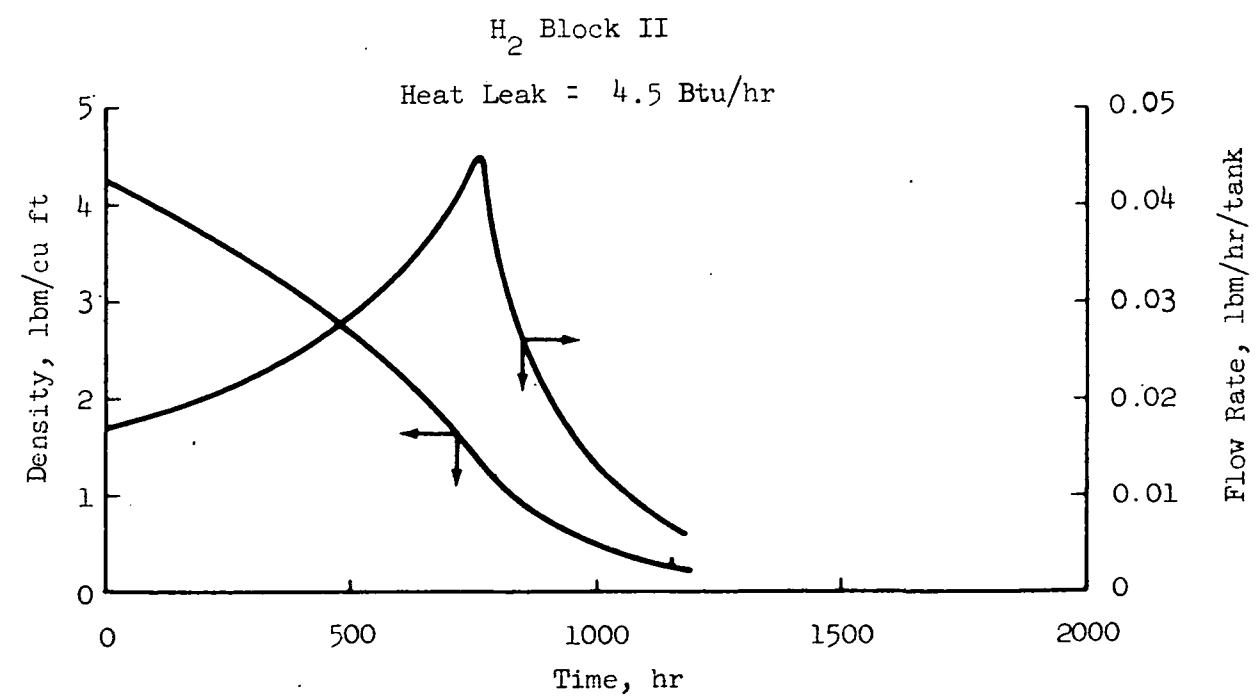
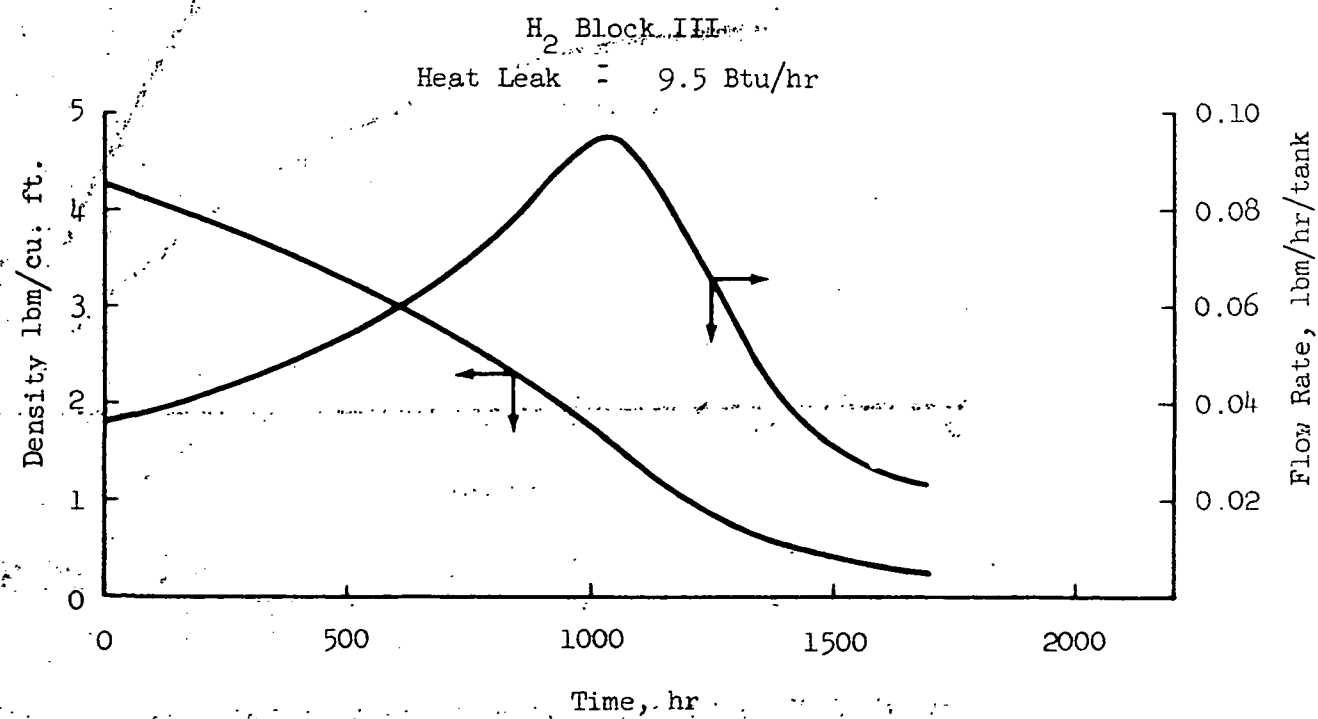
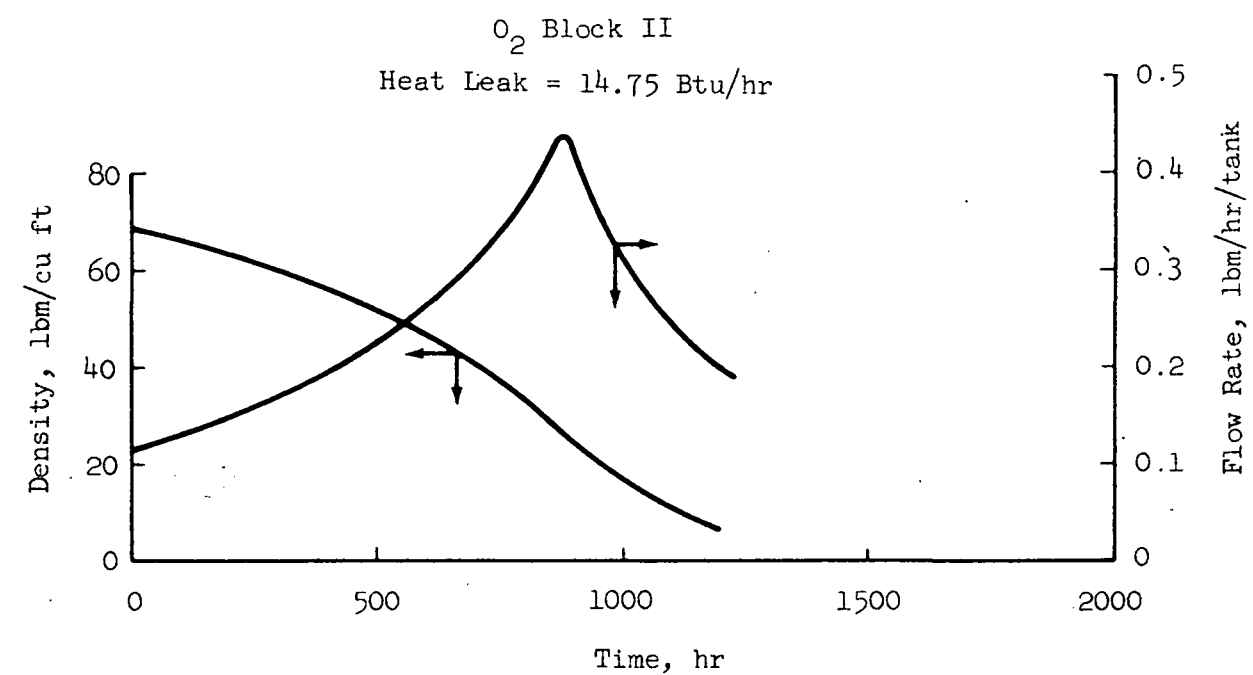
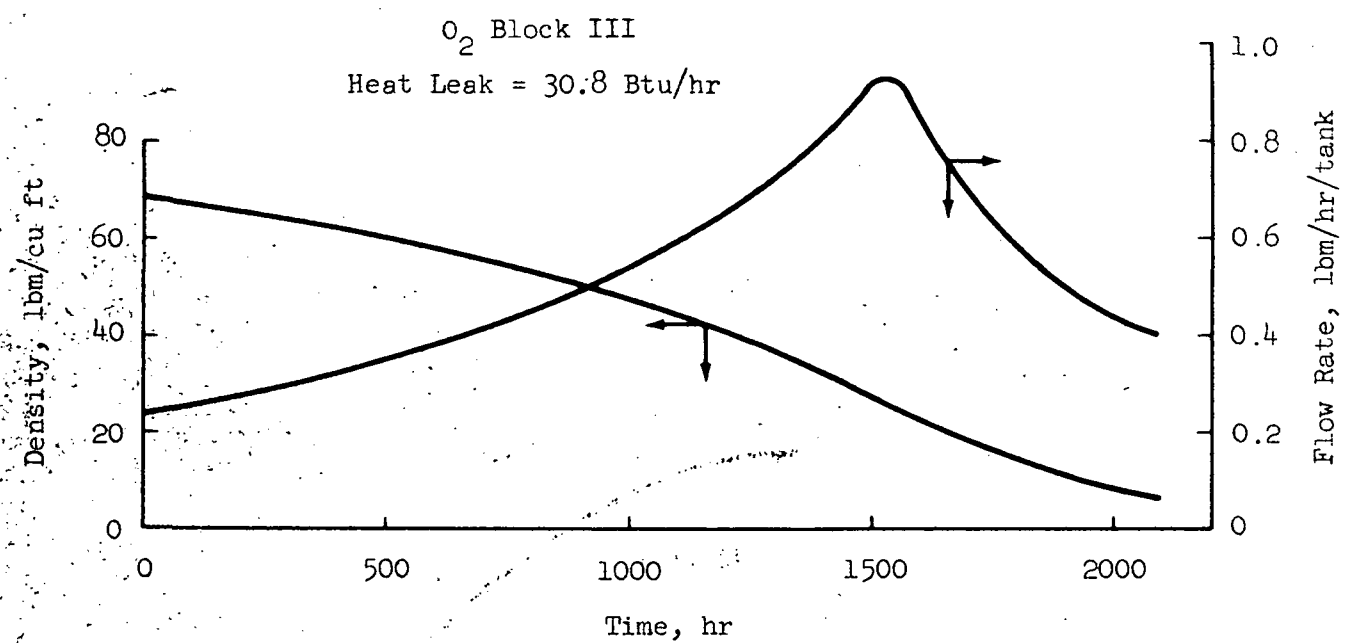


Fig. 4.1-1 Performance Characteristics of Apollo Reactant Tanks

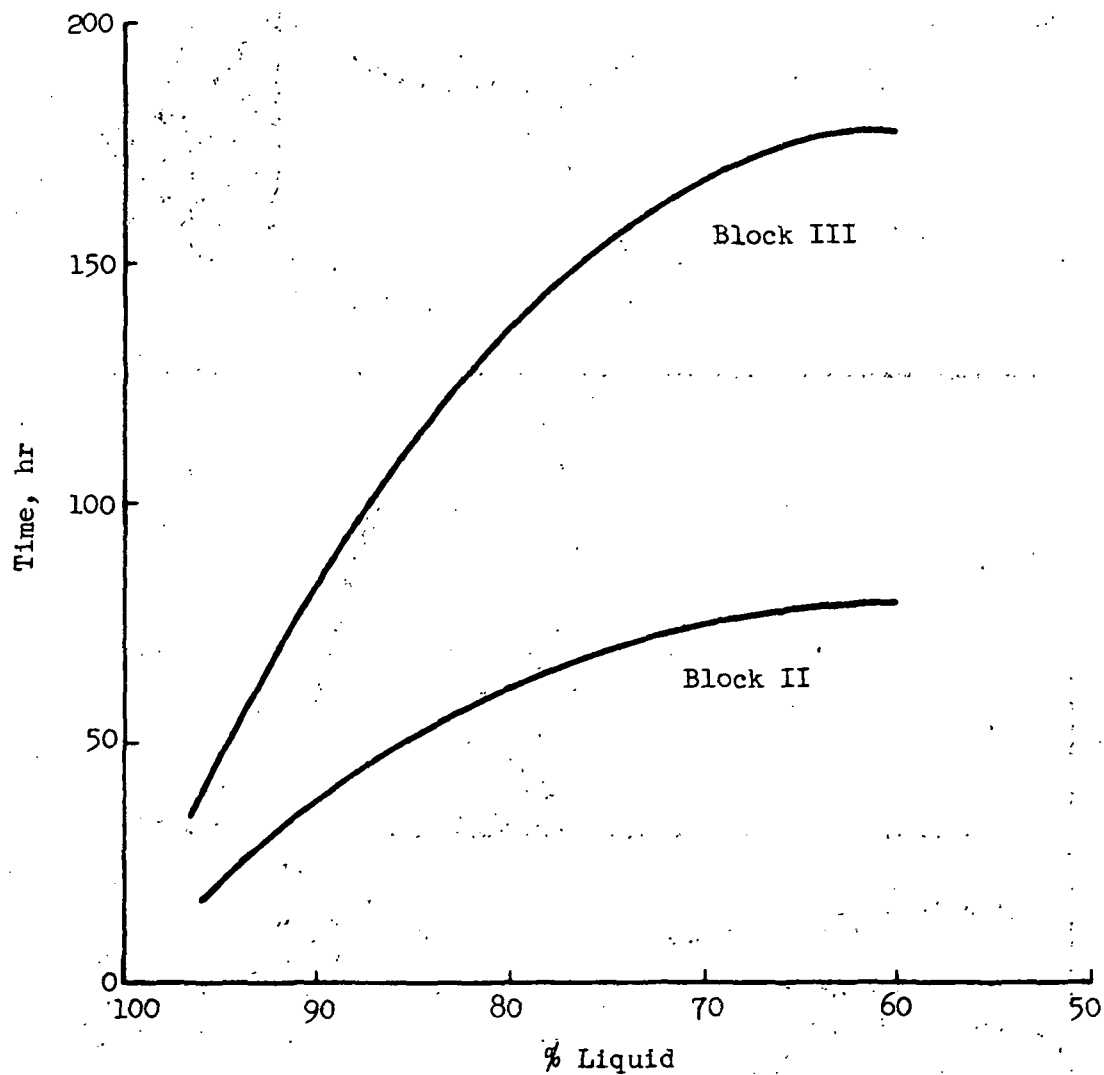
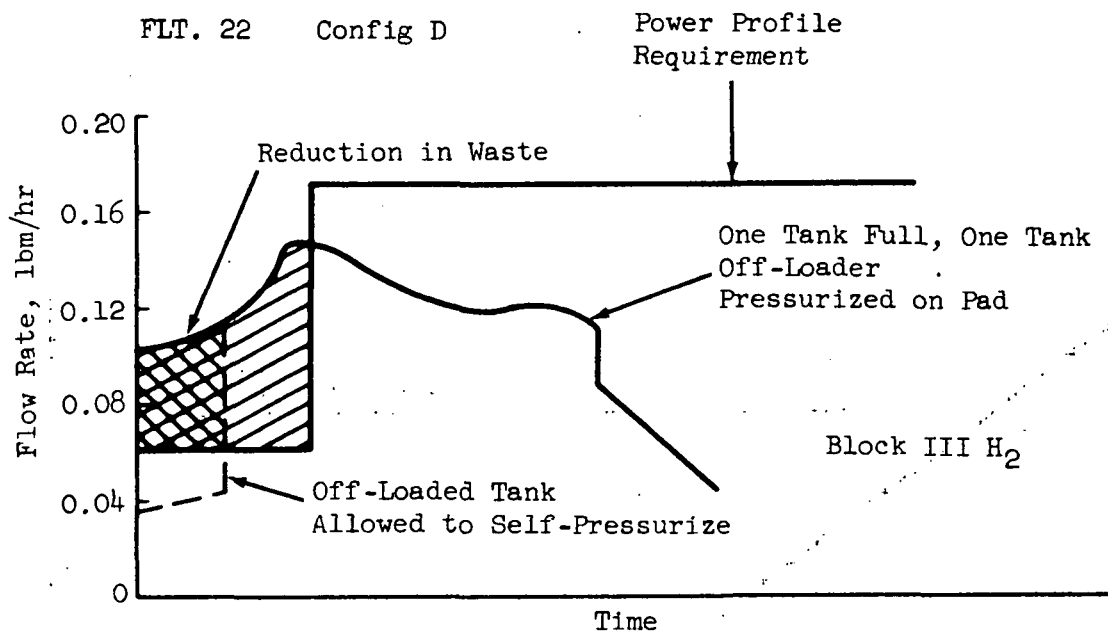


Fig. 4.1-2 Time to Reach Operating Pressure vs. % Liquid at Fill; Beech Block II & III H_2 Storage Tanks

A. Example of Self Pressurization
To Reduce Waste



B. Example of Selective Loading
to Eliminate Waste

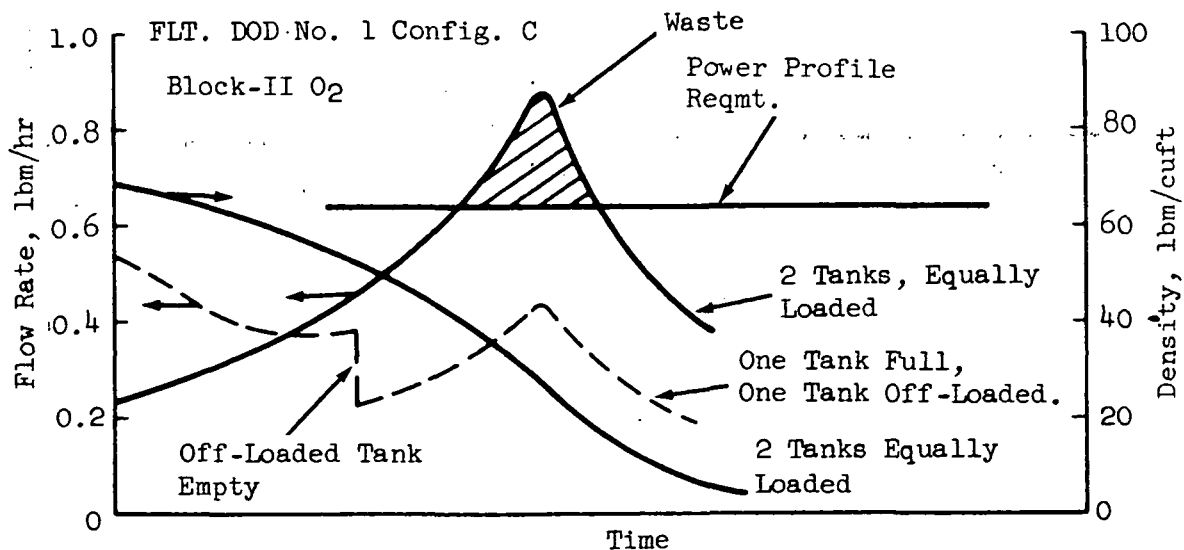


Fig.4.1-3 Typical Examples of Waste Calculations

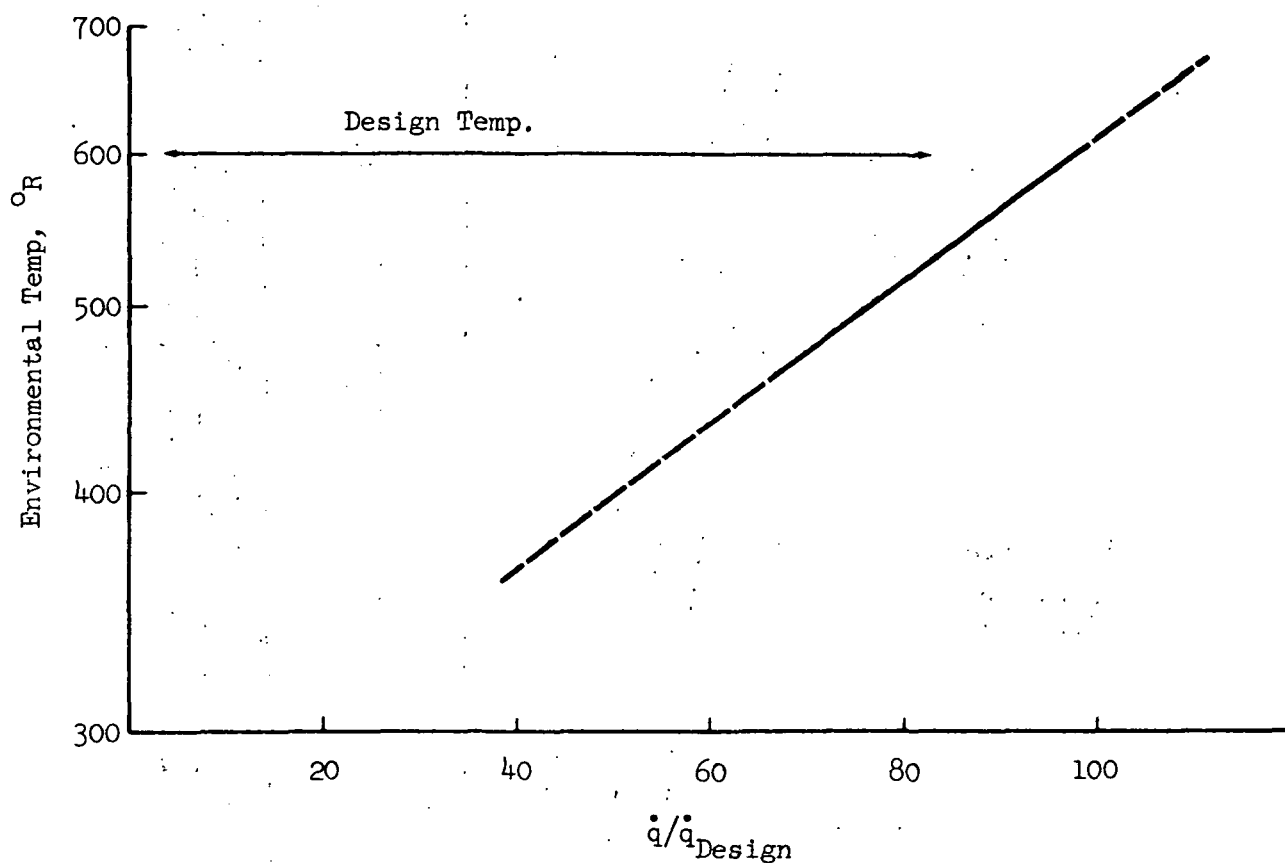


Fig. 4.1-4 Heat Leak vs. Environmental Temperature; Beech Block II & III H₂ Storage Tanks

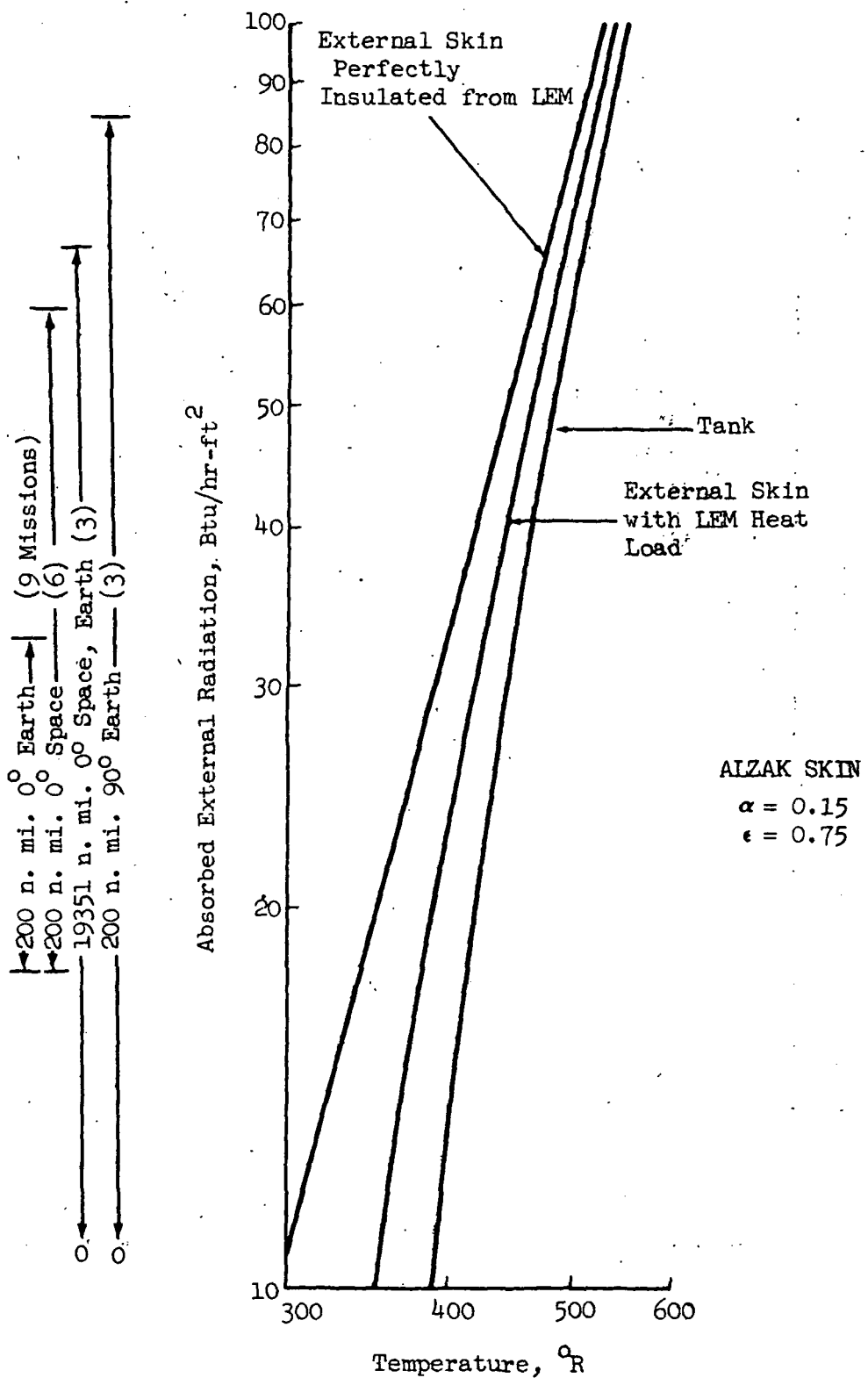


Fig. 4.1-5 Cryogenic Tank Compartment Temperature

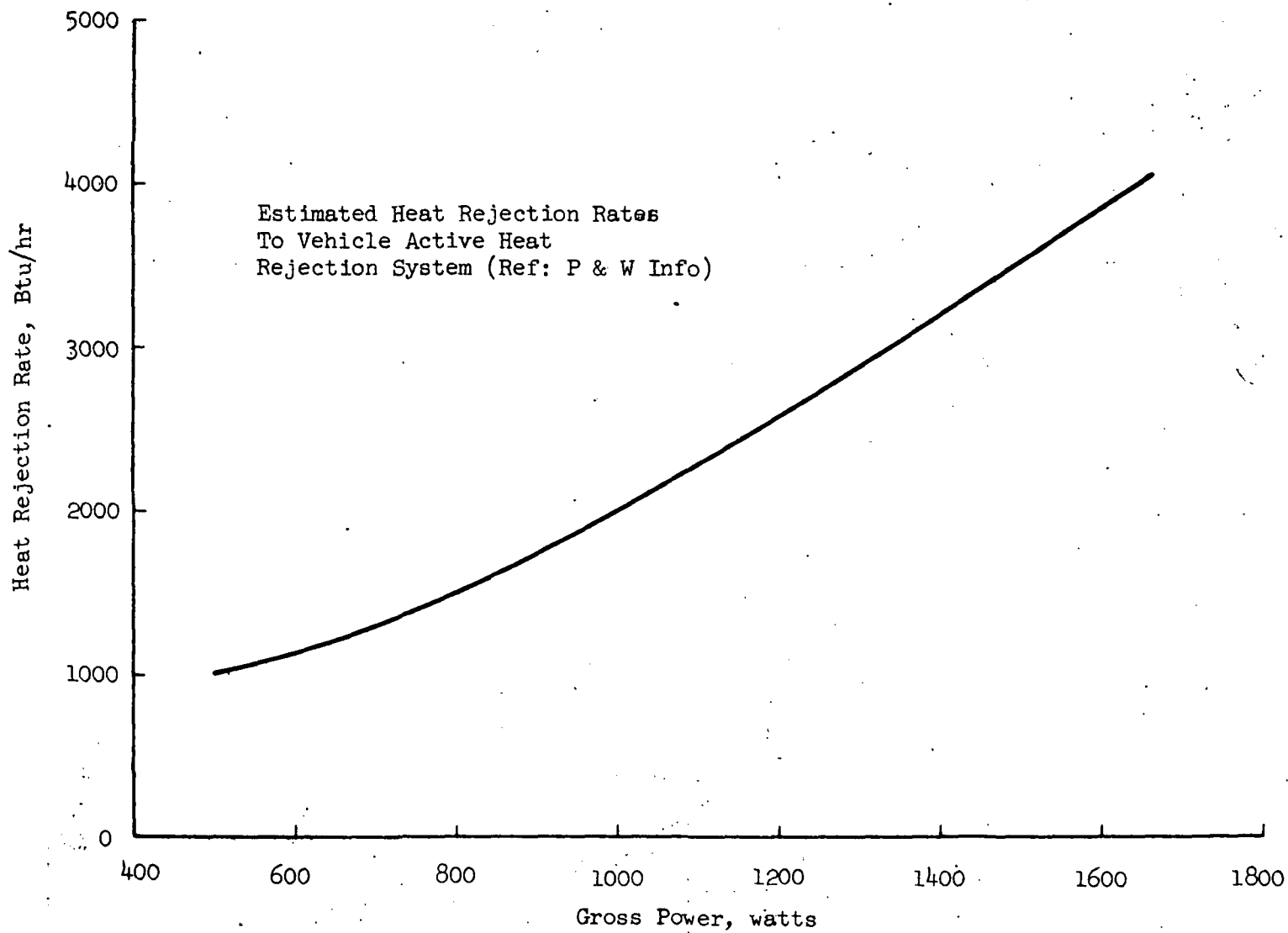


Fig. 4.1-6 Present Apollo CSM Powerplant

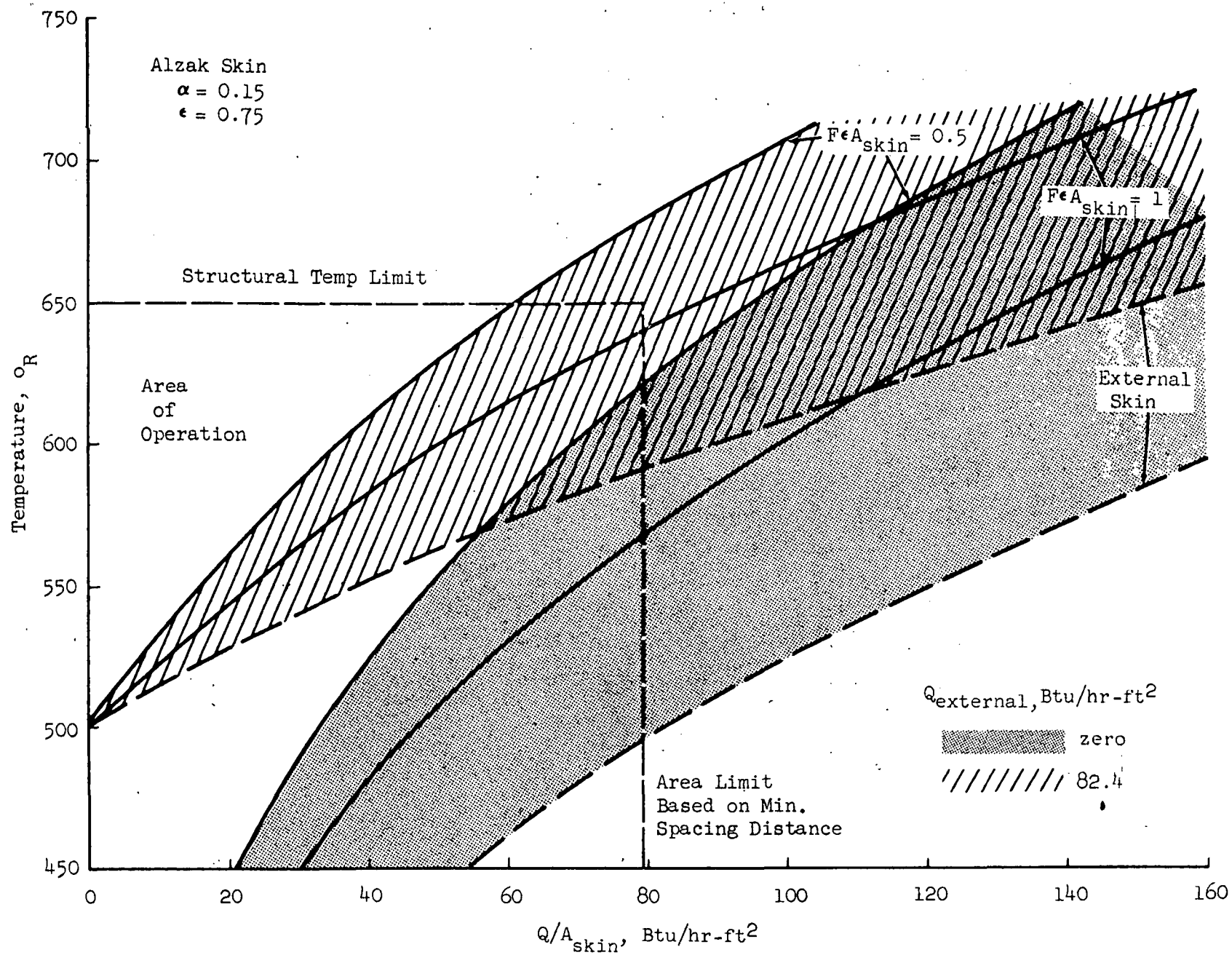


Fig. 4.1-7 Fuel Cell External Temperature

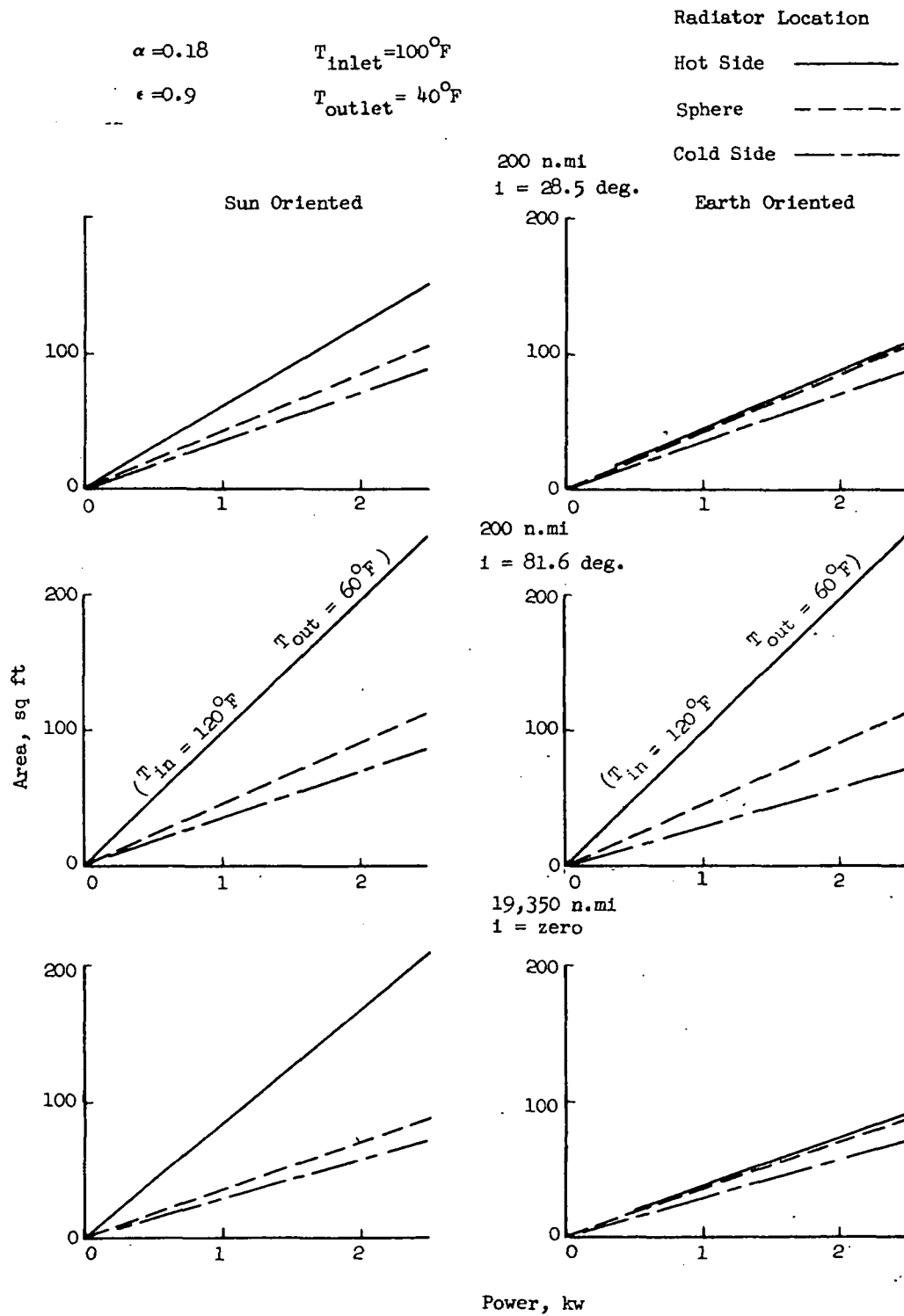


Fig. 4.1-8 Equipment Radiator Size vs. Power

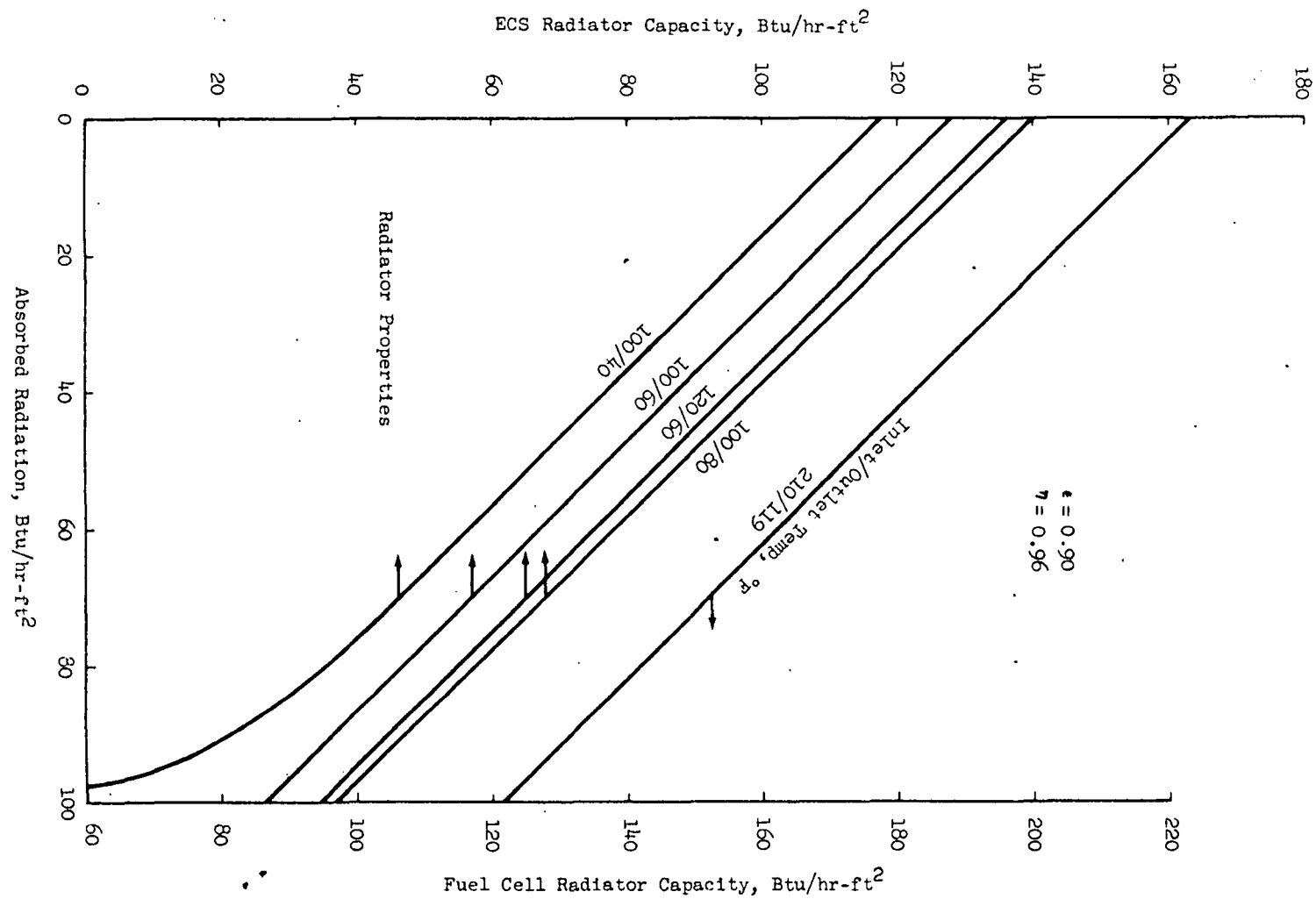


Fig. 4.1-9 Radiator Capacity vs. Environmental Radiator

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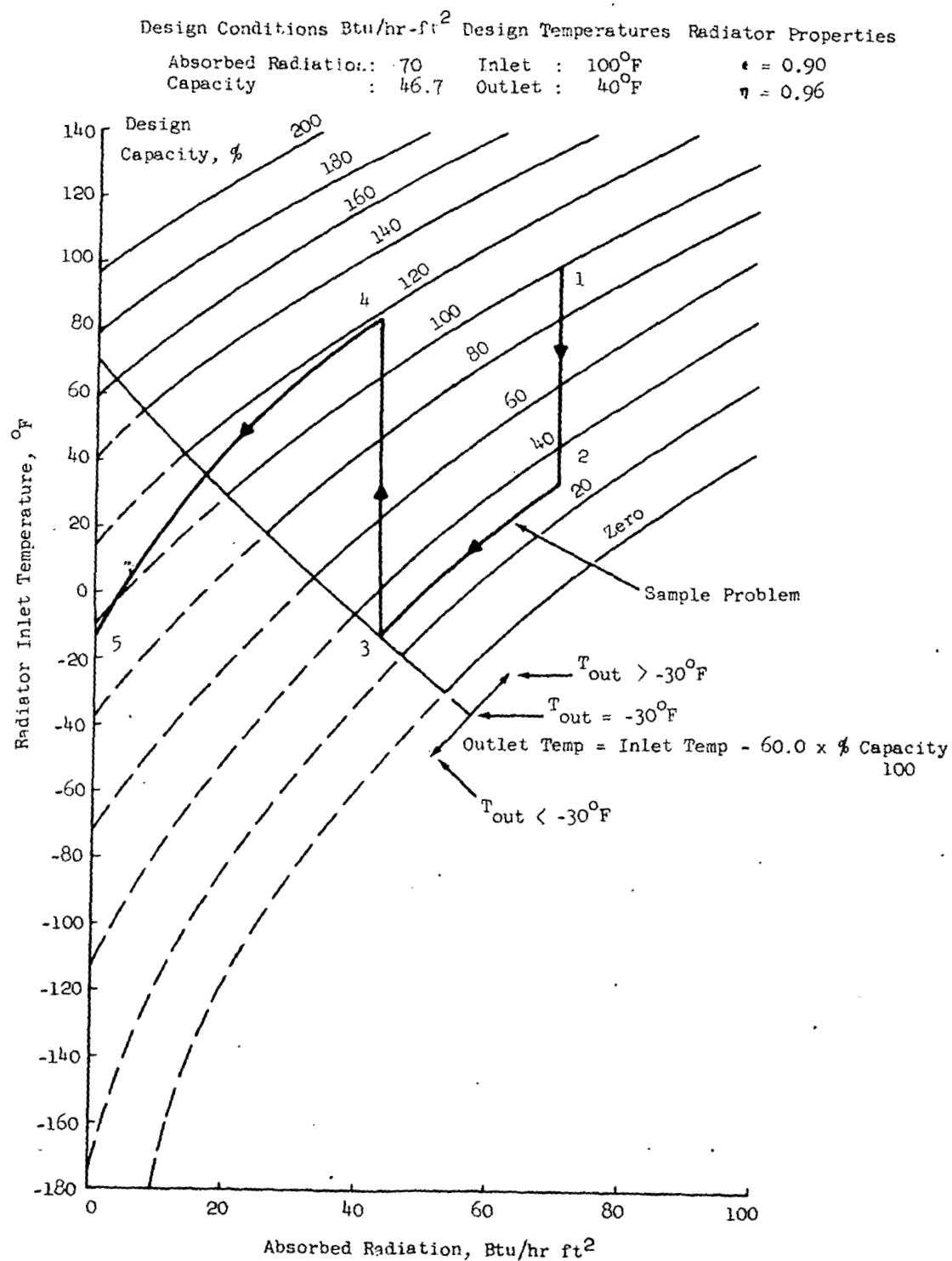


Fig. 4.1-10 ECS Radiator Performance

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4.1-25

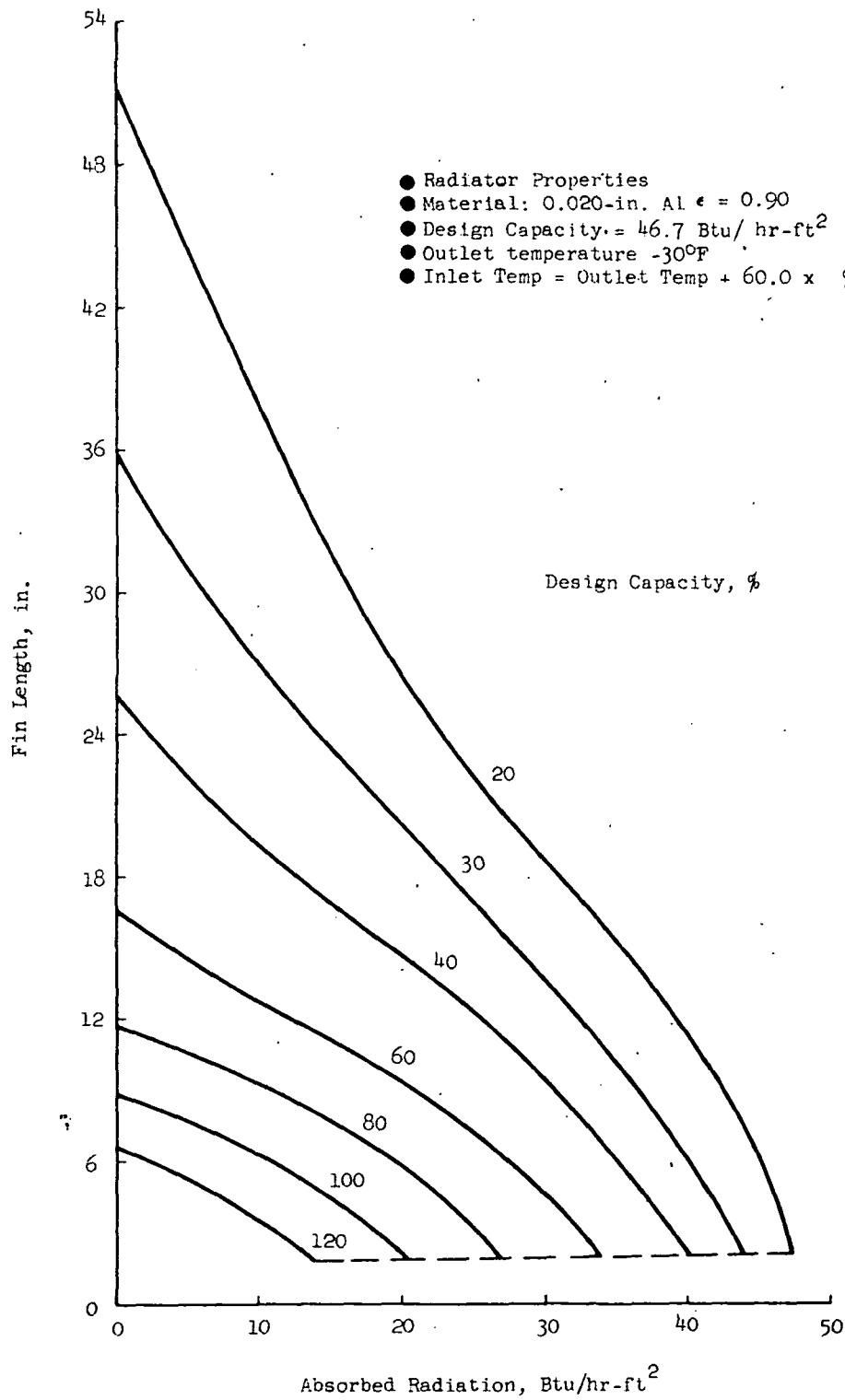


Fig. 4.1-11 Fin Length vs. ECS Radiator Capacity

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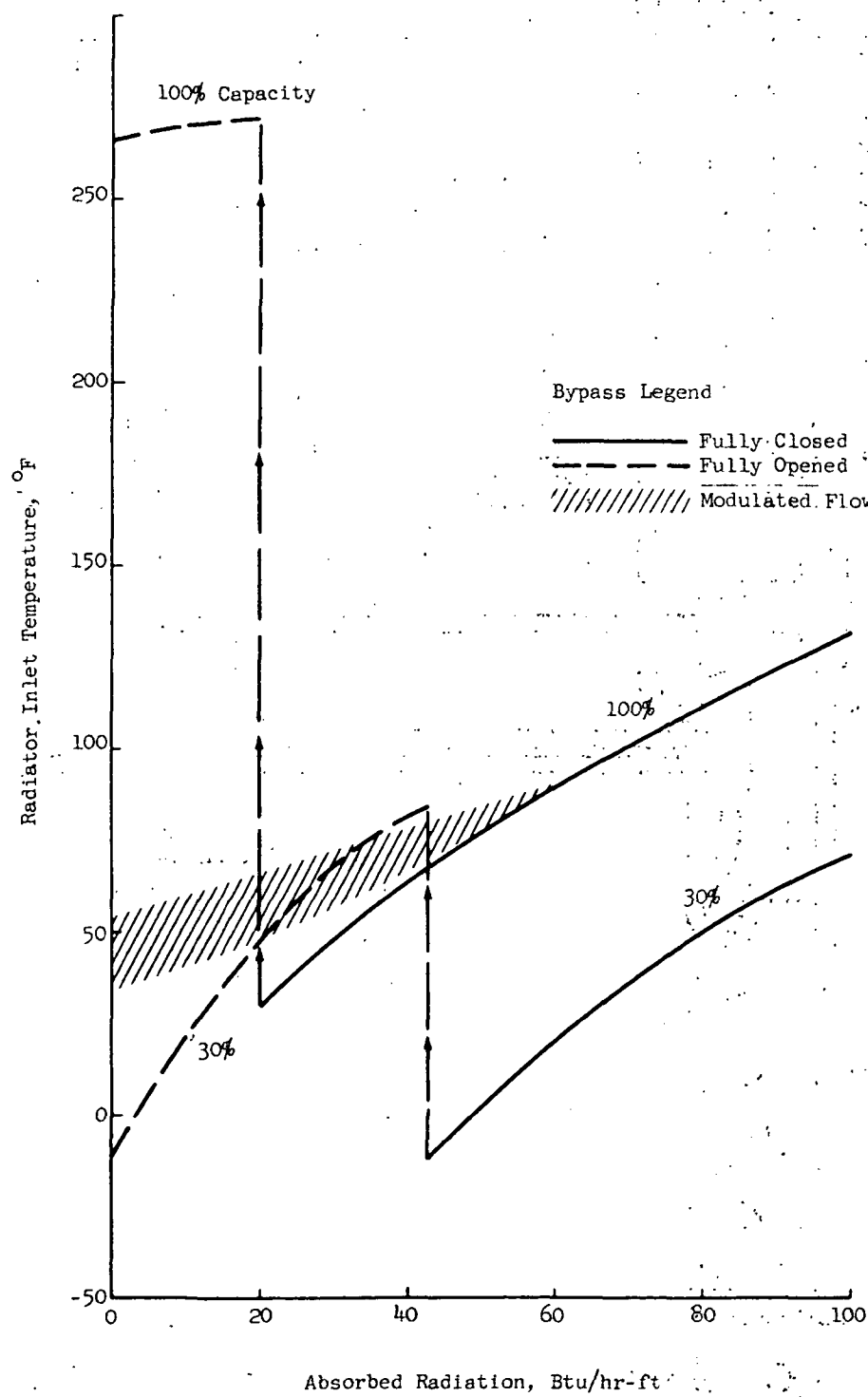


Fig. 4.1-12 Bypass Radiator Performance

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NO. 1
Bypass Control

NO. 2

NO. N

4.1-27

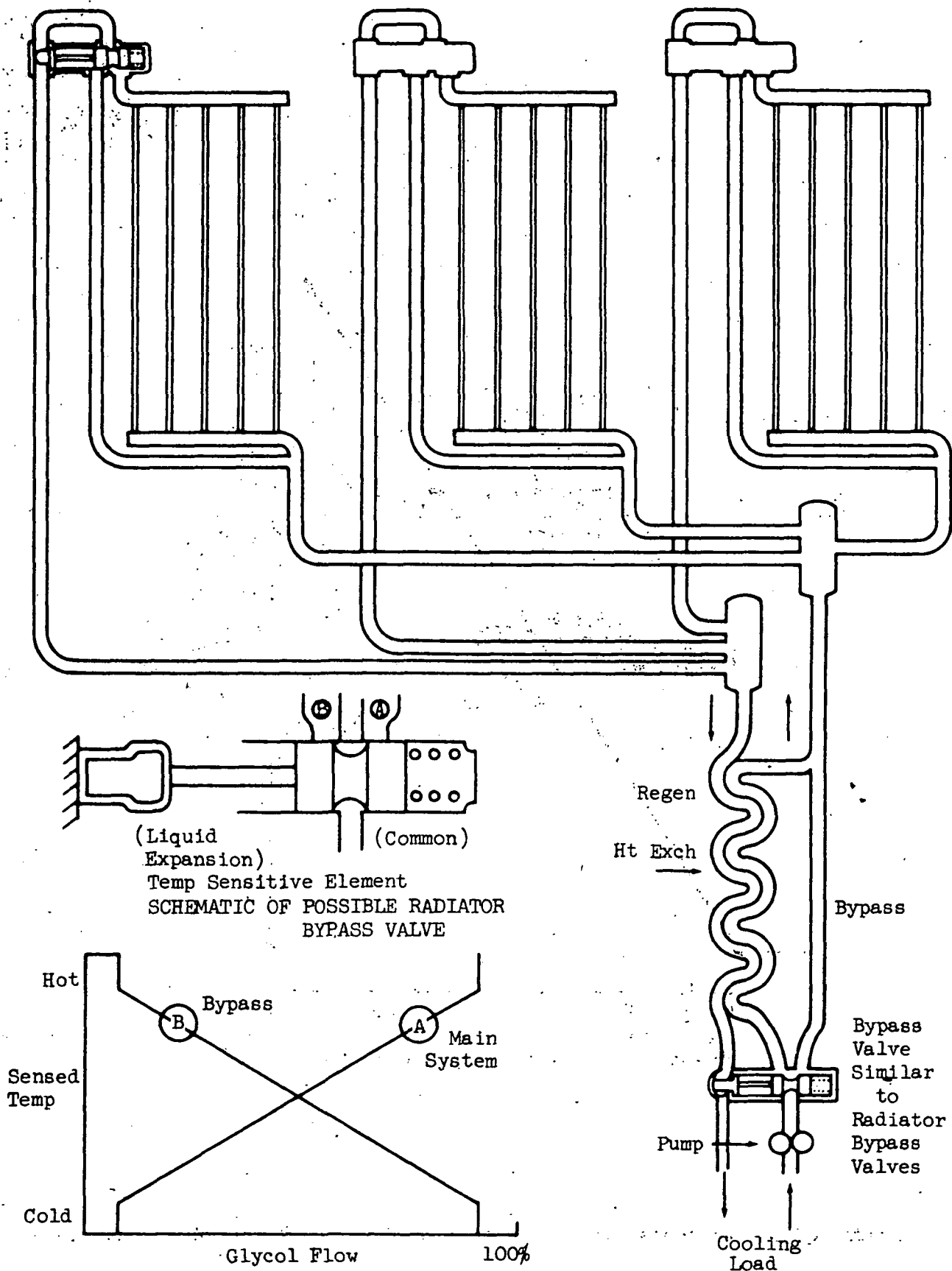


Fig 4.1-13 Schematic-Multiple Parallel Radiator System with Load Compensation

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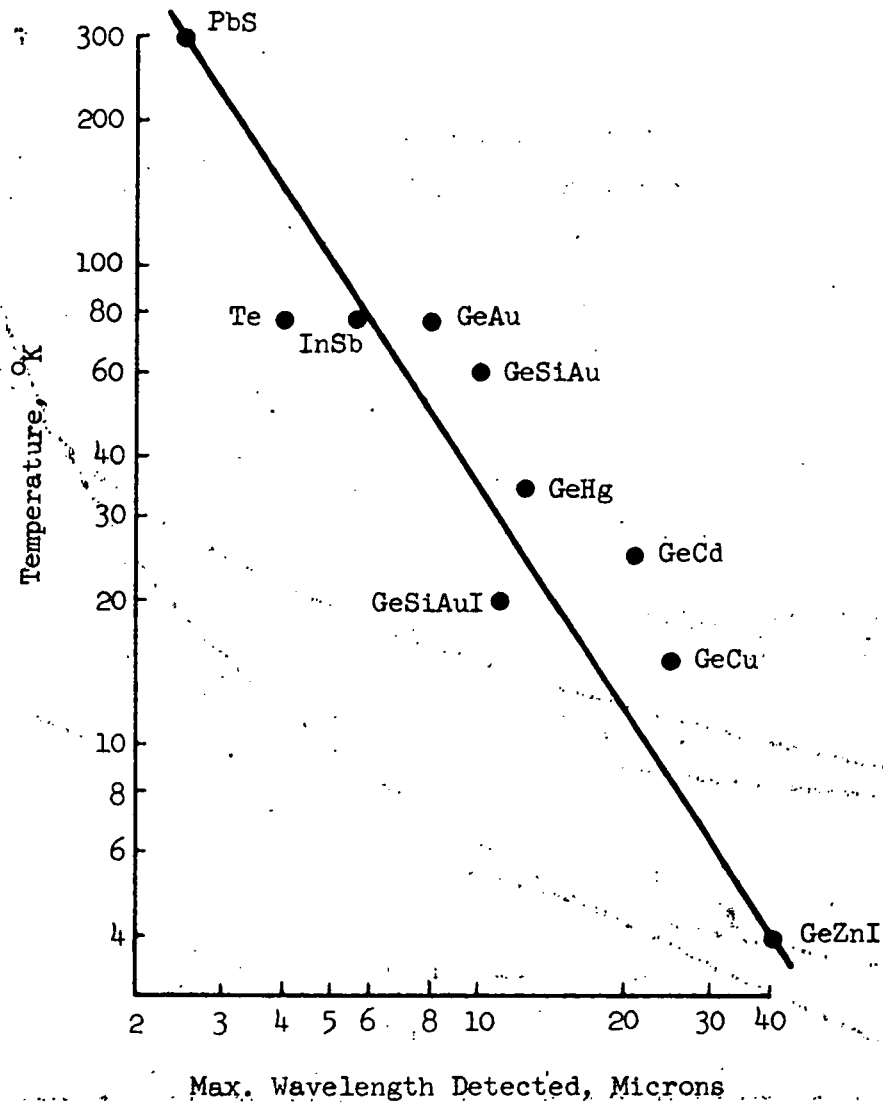


Fig. 4.1-14 Survey Plot of Effective Range of IR Detectors vs. Operating Temperature.

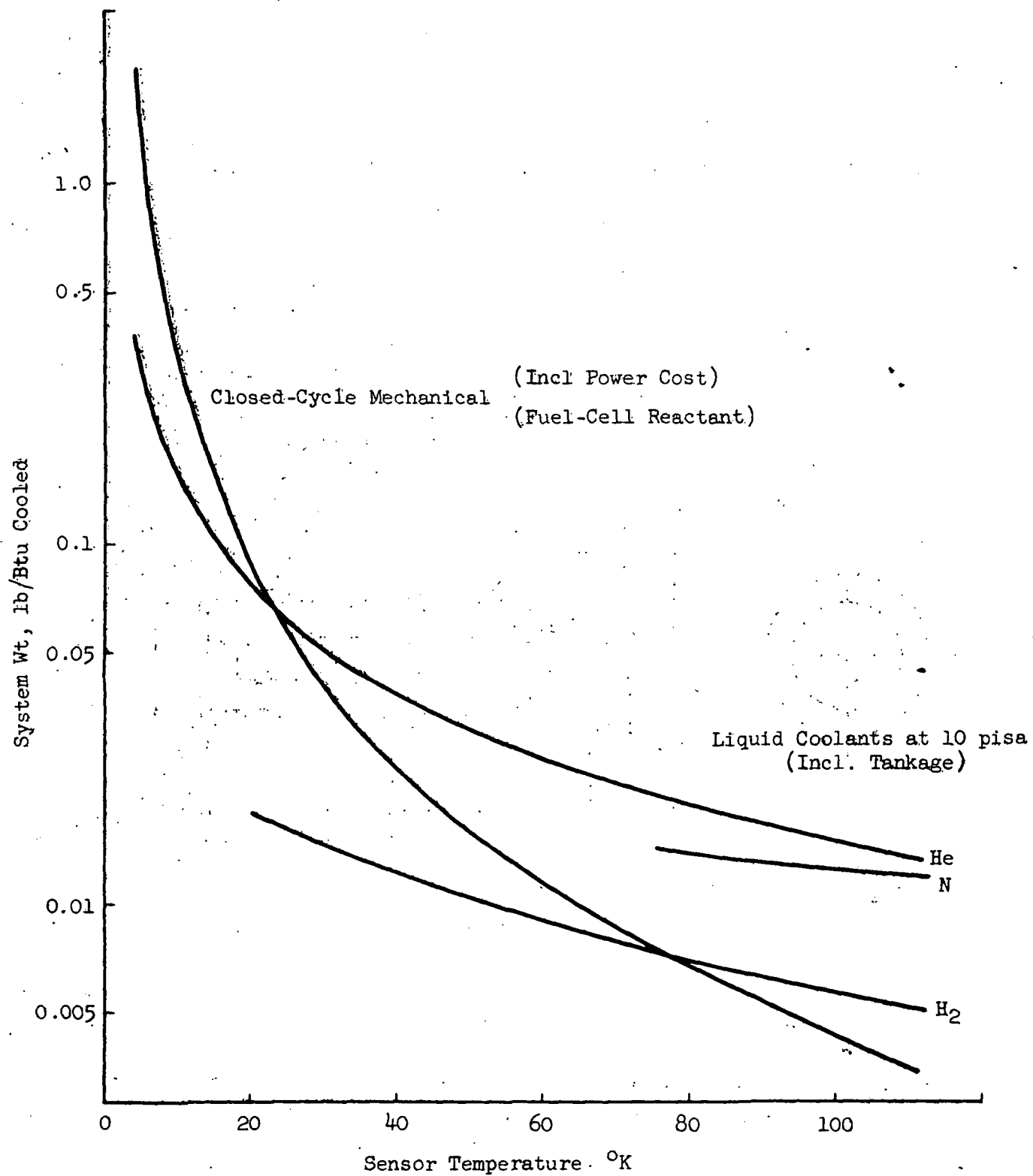


Fig. 4.1-15 Comparison of Storage and Mechanical Refrigeration Systems for IR Sensor Cooling (Typ. for 22-day Mission)

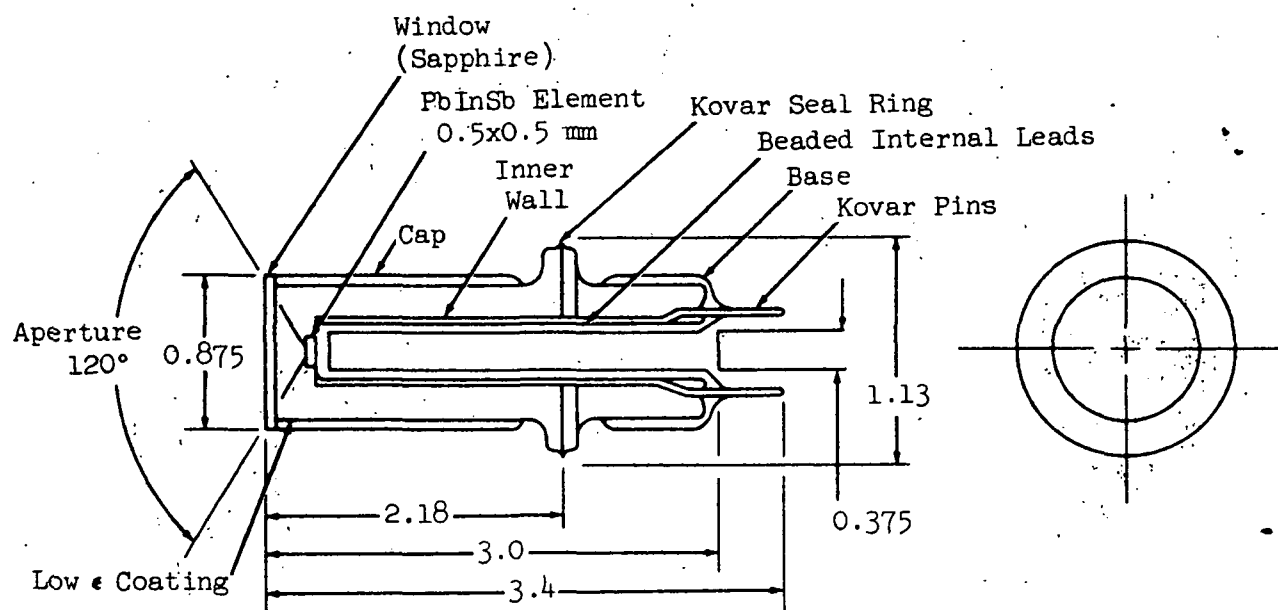


Fig. 4.1-16 Example of IR Detector (Honeywell LG67)

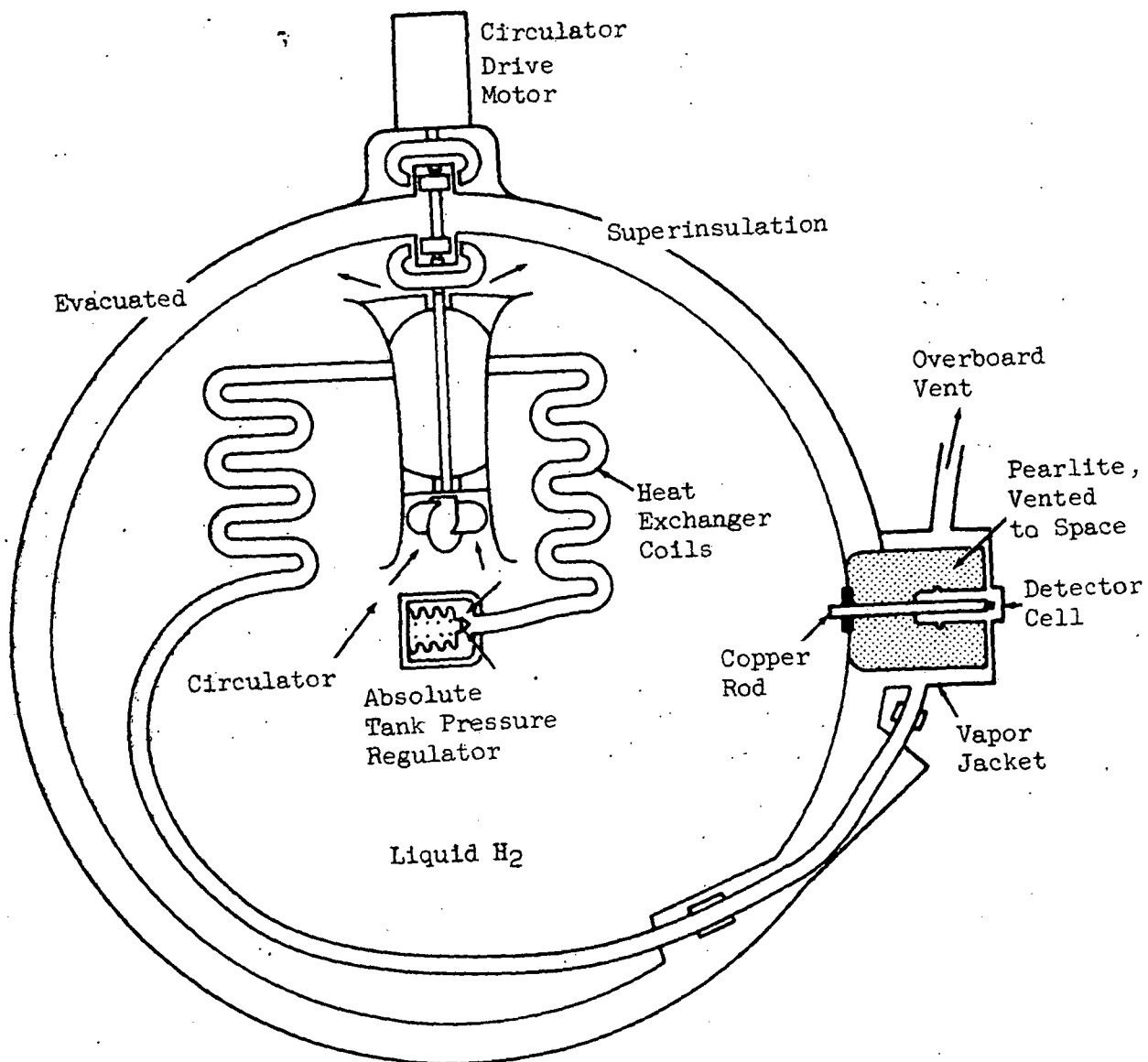


Fig. 4.1-17 Schematic-Possible Arrangement of Liquid H₂ Tank for IR Detector Cooling

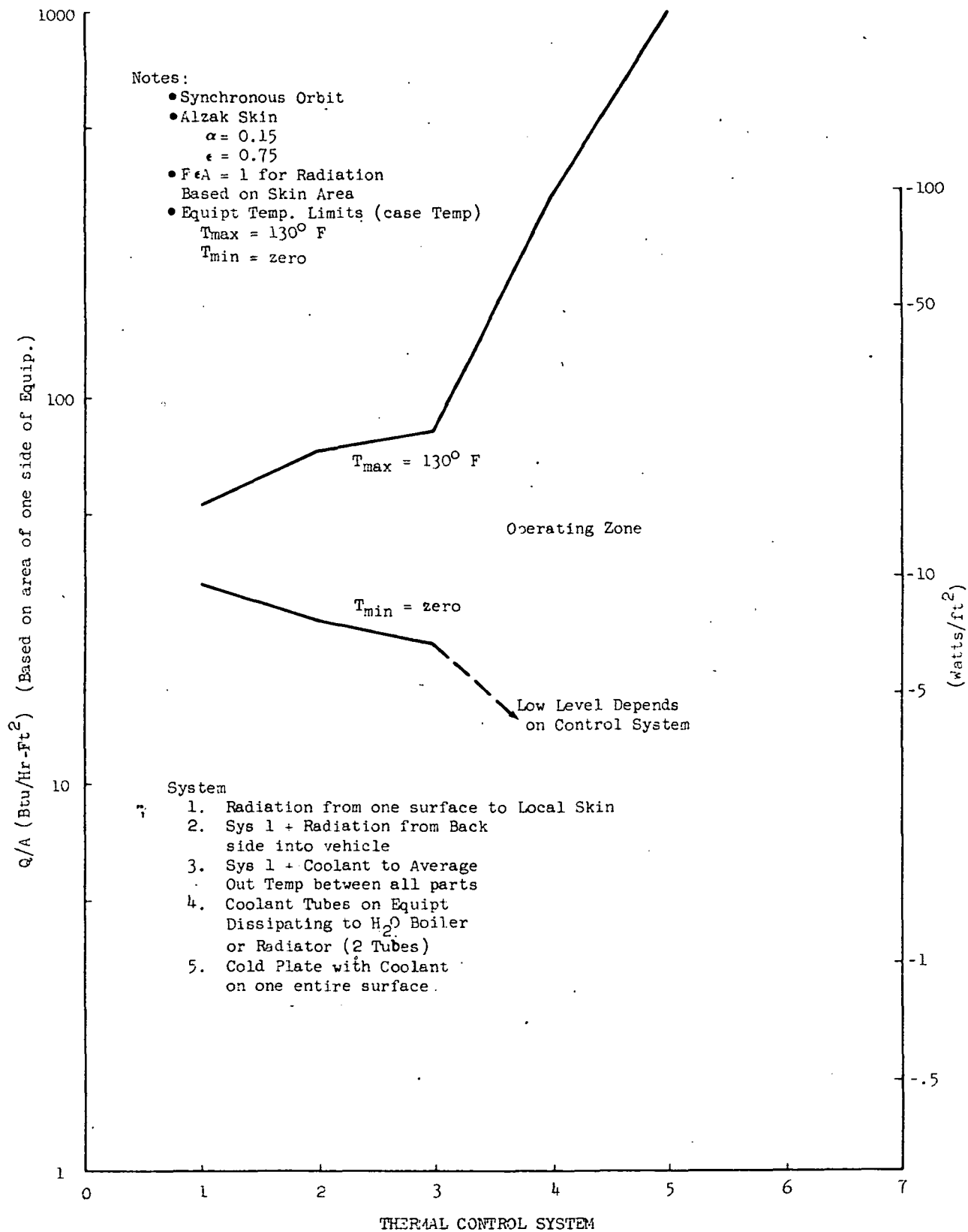


Fig. 4.1-18 Comparison of Thermal Control System's Capacity

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Table 4.1-1 CRYOGENIC TANKAGE SUMMARY

Flt No.	No. Tanks Req'd				Usable lbm		Waste lbm		Total lbm		Wet System Wt, lbm		No Waste Environ Temp °R	Obtainable Environ Temp, °R	Mod Pwr Profile Req'd	Hard Mount
	II	III	II	III	H ₂	O ₂	H ₂	O ₂	H ₂	O ₂	H ₂	O ₂				
209			1		0	225	0	0	0	231.4	0	321.1	610	490-590	No	Yes
211C		1		1	76	691	0	0	78.5	703	250	955.5	610	490-590	No	Yes
211D	1	1	1	1	125	1374	0	0	128.7	1392.4	389.4	1734.6	610	490-590	No	Yes
507	2		2		38.3	371	0	0	40.7	383.8	219.1	563.2	610	490-590	No	Yes
509	1		1		20.8	175	0	0	22	181.4	111.1	271.1	610	490-590	No	Yes
215	2		2		52.4	471	0	0	54.8	483.8	233.2	663.2	610	490-590	No	Yes
216C		1		1	56	475	22	275	80.5	762	252	1014.5	416	410-497	?	No
216D		2		2	132	1743	12	71	149	1838	492	2343	395	410-497	Yes	No
219C		1	1	1	75	1303	9.2	0	86.7	1321.4	258.2	1663.6	482	410-497	?	No
219D		2	1	2	156	2637	11	0	172	2667.4	575	3262.1	395	410-497	Yes	No
513	2		2		39.4	425.5	0	0	41.9	438.3	220.3	617.7	610	490-590	No	Yes
221C		1		1	67	1094	12.7	.7	82.2	1106.7	253.7	1357.2	455	413-452	No	Yes
221D		2		2	149	1691	12.0	0	166	1715	509	2220	422	413-452	?	No
516C		1		1	97.2	969	0	22	100.7	1003	272.5	1255.5	610	490-590	No	Yes
516D		2		2	205	2311	0	0	210	2335	553	2840	610	490-590	No	Yes
518C		1		1	93	856	1	35	96.7	903	286.3	1155.5	562	352-535	No	No
518D		2		2	180	2209	6	14	191	2247	534	2752	515	352-535	?	No
521C		1	2		58.4	545	19.3	209	80.2	766	251.7	1018	380	413-452	Yes	No
521D		2		2	139	1884	28	147	172	2055	515	2560	395	413-452	Yes	No
523C		1		1	80	739	6.5	80	89	831	260.5	1083.5	507	410-497	No	No
523D		2		2	166	2075	16	36	187	2135	530	2640	415	410-497	?	No

TABLE 4.1-1 CRYOGENIC TANKAGE SUMMARY (Cont.)

Flt. No.	No. Tanks Req'd				Usable lbm		Waste lbm		Total lbm		Wet System Wt, lbm		No Waste Environ Temp °R	Obtainable Environ Temp, °R	Mod Pwr Profile Req'd	Hard Mount
	H ₂	O ₂	H ₂	O ₂	H ₂	O ₂	H ₂	O ₂	H ₂	O ₂	H ₂	O ₂				
229C		1		1	99.4	906	.4	18.4	102.3	936.4	273.8	1188.9	595	410-497	No	Yes
229D		2		2	187.8	2238	5.0	5.0	197.8	2267	540.8	2772	541	410-497	No	No
230C	1	1		1	119	1000	0	0	122.7	1012	383.4	1264.5	610	490-590	No	Yes
230D		2		2	204	2363	0	0	209	2387	552	2892	610	490-590	No	Yes
DOD1C		1		1	46.3	450	12.3	50	61.1	506.4	232.6	758.9	460	413-452	No	No
DOD1D		1		1	86.8	1100	0.5	0	90.8	1112	261.3	1364.5	580	413-452	No	No
DOD2C	1	1		1	118	1010	0	0	121.7	1022	382.4	1274.5	610	490-590	No	Yes
DOD2D		2	2	1	161	1688	0	0	166	1713	509	2145	610	490-590	No	Yes
DOD3C		1	2		46.1	396	10.5	0	59.1	408.8	230.6	588.2	460	413-452	No	No
DOD3D		1		1	85.2	1054	.4	0	88.1	1066	256.7	1318.5	560	413-452	No	No
DOD4	1		1		18.5	240	0	0	19.7	246.4	108.9	336.1	610	490-590	No	Yes
DOD5	1		1		20	297	0	0	21.2	303.4	110.4	393.4	610	490-590	No	Yes
DOD6	2		2		43	442.5	0	0	45.5	455.3	223.9	634.7	610	490-590	No	Yes

TABLE 4.1-2 ORBITAL CHARACTERISTICS FOR THERMAL DESIGN

Flt. No.	Alt, n.mi	Orbit Inclination		Duration, days	Orientation			
		To Equator	To Solar Vector		Flight Direction	Earth Facing	Ecliptic Plane Facing	Sun Facing
			min. max.					
209	200	28.5	Zero	52	14	-X	+Z	
211C 211D	200 200	28.5		52	30	+Y	+Z	
507	200	81.5		90	14	+X	+Z	
509	19,351	0		23.5	14			Random
215	200	50		73.5	14	-X	+Z	
218C 218D	200	28.5		52	45	(218 LEM +X Docks to 219-X)		Random
219C 219D	200	28.5		52	45			Random
513	200	81.5 +1 deg/day		90	14	-X	+Z	
221C 221D	200	28.5		52	45	Spin in Orbit Plane 3-9 rpm ± X-Axis		
516C 516D	19,351	0		23.5	45			Random
518C 518D	200	81.5 (-1 deg/day Retrograde Orbit	90	90	45	+X	+Z	
521C 521D	19,351	0	Zero	23.5	45	+Y	+Z	
523C 523D	200	28.5		52	45			Random +X to 229 230
229C 229D	200	28.5		52	45			Random -X to 523
230C 230D	200	28.5		52	45			Random -X to 523
DOD1C 1D	160 160	32		55.5	30			Random
DOD2C 2D	160 160	32		55.5	30	±X	+Z	- -
DOD3C 3D	160 160	32		55.5	30	±Y	+Z	- -
DOD 4	160	32		55.5	14	±Z	-X	- -
DOD 5	160	32		55.5	14	±Z	-X	- -
DOD 6	160	32		55.5	14	±Z	-X	- -

TABLE 4.1-3 THERMAL DESIGN SUMMARY TABLE

Flt. No.	Avg. LEM Power, Watts	Peak LEM Power Over Avg.	Avg. Fuel Cell Power, Watts	Water Gen, lb	Eqipt Size, sq ft	Radiator Location in Desc Stage	H ₂ O Boiler Reqmts for Eqipt, lb	Fuel Cell Size, sq ft	Radiator Location (A:Asc,D:Desc)	H ₂ O Boiler Reqmts for FCA, lb	H ₂ O Req'd For Exper, lb	Total H ₂ O Req'd, lb	H ₂ O Avail for other use, lb
209	433w x 336 hrs 146 kw hr		(Assume 56.5w is lost to Structure)				340	0	-	0	58	398	0
209 Alter-mate	355	~ 78 w cont	0	0	(21)		0	0	-	0	58	58	340
211C	1021		1279	756	42 = (21+21)	I, III		15.1		4.7		151.	605.
211D	1021-1099	4049w/1 hr, 10x/Mission	1257-2593	1182	46 = (23+23)	I, III	138	30.8 = (15.4+15.4)	A(+X, -Z)	8.8		155	1027
507	1305	5400w/3 min, 75x/Mission 1200w/48 min, 102x/Mission	1447	367	70 = (35+35)	II, III	325	29.2 = (14.6+14.6)	A(+X, -Z)	0	42	367	0
509	550	369w/3 hr, once	786	195.	20 = (10+10)	II, III	3.8	10	A(+X, -Z)	2.1	13.2	19	176.
215	1465	595w/45 min, 214x/Mission	1937	502	60 = (30+30)	II, III	325	20.6 = (10.3+10.3)	A(+X, -Z)	138	13.2	476.	26.
218C 218D	392 392-480	None	628 650-1964	531 1273	22 = (11+11) 24 = (12+12)	II, III	0	7.53 26.2 = (13.1+13.1)	D(II) D(II, IV)	0	14.3	14	517. 1259.
219C 219D	622 622-700	None	880 880-2194	723 1499	28 = (14+14) 30 = (15+15)	I, II, III	0	11.3 30.6 = (15.3+15.3)	A(+X, -Z)	0	805	805	-82 694
513	1078	492/40 min, 13x/Mission	1314	360	48 = (24+24)	II, -X	14.5	9.8	A(+X, -Z)	9.2	575	81.	279.
221C 221D	549 (+1200 Btu/hr Crew)	27w/4-5x/day for 18 hr total	785 785-2099	652 1430	38 = (19+19)	I, III	1.66	7.85 23.2 = (11.6+11.6)	A(+X, -Z)	1.26	8.8	12. 12.	640. 1418.
516C 516D	860 860-938	None	1096 1118-2432	940 1793	38 = (19+19) 40 = (20+20)	II, III	0	15.6 36.2 = (18.1+18.1)	A(+X, -Z)	0	41	41	697 1752
518C 518D	916 916-994	1950w/3/4 hr, 8x/day, 45 day 650w/3/4 hr, 3x/day, 22 day	1048 1070-2384	893 1721	60 = (30+30) 64 = (32+32)	II, III	730	14.5 29 = (14.5+14.5)	A(-Z)	0	154	884	9 837.
521C 521D	450 450-500	800w/1 hr, Once	669 669-1983	554 1340	18 = (9+9) 20 = (10+10)	II, III	2.7	8.56 27.4 = (13.7+13.7)	A(+Y, +X)	2.0	26.4	31. 31.	523. 1309.
523C 523D	539 539-617	305w/1.25 hrs, 3x/Mission	929 959-2265	770 1590	24 = (12+12) 26 = (13+13)	II, III	3.9	12.2 31.8 = (15.9+15.9)	D(IV) D(IV, +Z)	3.3	52	59 59	711 1531

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TABLE 4.1-3 THERMAL DESIGN SUMMARY TABLE (Cont.)

Flt. No.	Avg. LEM Power, Watts	Peak LEM Power Over Avg.	Avg. Fuel Cell Power, Watts	Water Gen, lb	Eqipt Size, sq ft	Radiator Location In Desc Stage	H ₂ O Boiler Reqmts for Eqipt, lb	Fuel Cell Size, sq ft	Radiator Location (A:Asc,D:Desc)	H ₂ O Boiler Reqmts for PCA, lb	H ₂ O Reqd For Exper, lb	Total H ₂ O Reqd, lb	H ₂ O avail for other use, lb
22JC 22JD	872 872-950	762w/5 min, 5x/day	1108 1130-2444	963 1782	38=(19+19) 44=(22+22)	I,II,III	48.6	15.2 35.6= (17.8+17.8)	A(+X,-Z)	18.15 23.6	646	713 718	250 1064
23JC 23JD	1018 1018-1096	1020w/90 min, 3x/day	1254 1276-2590	1119 1965	46=(23+23) 48=(24+24)	I,III	704	18.5 38.8= (19.4+19.4)	D(IV) D(IV)	104 162	22	830 888	289 1077
DOD-1C DOD-1D	549	1180w/35 min, 60x/Mission	825 825-2139	450 837	26=(13+13)	I,III	142	10.5 28.4= (14.2+14.2)	A(-Z,+X)	60.2 82	7.7	210 232	240 605
DOD-2C DOD-2D	1565 1565-1643	1050w/6 hr/day	2015 2037-3351	1128 1546	64=(32+32) 66=(33+33)	I,III	645	21.8= (10.9+10.9) 37.5= 3x12.5	A(-Z,+X)	437 606	46	1128 1299	0 247
DOD-3C DOD-3D	575 575-653	1344w/1/2 hr, 1x/day 2450w/5 min, 1x/day 3200w/1/2 min, 1x/day	811 811-2125	442 836	24=(12+12) 26=(13+13)	I,III	92.5	8.6 23.6= (11.8+11.8)	A(-Z,+X)	31.3 38.7	7.7	132 139	310 697
DOD 4	433	462w/6 hrs, 1x/day	691	176	28=(14+14)	II,III	61	13.1	A(-Z,+X)	0	115	176	0
DOD 5	448	395w/6 hrs, 1x/day	734	188	30=(15+15)	II,III	37	13.4	A(-Z,+X)	0	15	136	0
DOD 6	1160	20kw/1/2 hr, 7x/Mission 5kw/1/2 hr, 14x/Mission	1632	419	90=(30+30+30)	II,III,IV	335	35.4= (17.7+17.7)	A(-Z,+X)	0	34	419	0

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TABLE 4.1-4 SPECIAL THERMAL CONTROL

Flt. No.	Cryogenic Cooling I-R Detectors			Active Heaters + Insul. Telescopes & Cameras			Passive Control Antennas	Miscellaneous	
	Exper	Watts	lb	Exper	Watts	lb			
209				0501 1501 1502 1503	6 6 . .	3 3 .			
211				0501	6	3			
507	0901	200	150	0501 0901 0902	6 30 3	3 60 5	1201 1202 *	1202	A separate glycol loop is required 150 lb for glycol, radiator, tanks, plumbing 50 watts pump power.
509				0501 0801	6 5	3 3		1501 1603	Specially-cooled equipment is located on CSM and is therefore not included in this study.
215	0901	200	150	0501 0901 0902	6 30 3	3 60 5			
218				0501	6	3			
219									
513	0802	200	150	0501	6	3		1602	Specially-cooled equipment is located on CSM and is therefore not included in this study.
221				0501	6	3			
516	0701	200	25	0501 0801 0702 0703 0704 0705A	6 5 15 10 10 20	3 3 85 30 10 5	0705 A *		
518	0901	200 (280 W Assumed By Exper)	150	0501 0901 0902 0903	6 30 3 5	3 60 5 5	0903 *		
521				0501 0801	6 5	3 3		1404	• Separate Glycol Loop & Radiator • Resistance Grids External • Zero Power Req'd. 200 lbs
523	0701	200	25	0501 0702 0703 0704 0705A	6 15 10 10 20	3 85 30 10 5	0705A *		
229									
230									
DOD 1									
DOD 2	Exp. 10	600 W. Cooler is Part of Exper.		Exp. 10	10	10			
DOD 3							Exp. 14 *		
DOD 4							Exp. 14 *		
DOD 5							Exp. 14 *		
DOD 6							Exp. 14 *		

* Use Coatings with $0.3 < \alpha/\epsilon < 1.0$ ~~CONFIDENTIAL~~

4.2 LEM LAB - RELIABILITY CONTROL

The study areas described herein are the preliminary mission design goals, subsystem reliability evaluation, spare equipment list, and a survey of the LEM/CSM backup potential.

4.2.1 Preliminary Mission Design Goals

In utilizing preliminary reliability objectives for earth-orbiting missions as a basis for design requirements the major emphasis has been (a) to evaluate the reliability of the subsystems to perform the missions, (b) to compare the obtainable reliability levels to the minimum mission objective, and (c) to recommend an approach to each subsystem configuration. Consideration has been given to minimizing the requirements for new development and qualification programs. It should be noted that no subsystem reliability apportionment has been attempted, since these apportionments are generally of an arbitrary nature and do not contribute to an optimum design approach. Rather, the configurations were studied until the full scope of the system problem became known, and then reliability improvements were applied where most advantageous and necessary. The system reliability was thereby reviewed to determine the areas offering the largest system reliability improvement.

It has been assumed that the overall goal for the probability of mission success (booster, CSM and LEM Lab) is 0.90. Based on Rept. No. 64-105, vol. 21 (Apollo - X Study, Contract NAS9-3140) the LEM Lab requirement is defined in the following table:

Mission Category	Total System	Booster	CSM	LEM Lab
Minimum Mission Success	0.90	0.95	0.95	0.988
Complete Mission Success	0.90	1	0.95	0.947

Minimum mission success is defined as the probability of not aborting the mission because of booster, CSM, or LEM Lab failure. Although such an event would cancel the scientific and engineering experiments, they could not be judged failures, since their exercise would be voided prior to any possibility of initiation.

Complete mission success includes experiment equipment reliability derived from booster success - upon which all the experiments are predicated. These preliminary mission design goals define the minimum subsystem reliabilities necessary to the mission requirements. It is obvious that no supporting subsystem can have a reliability level less than 0.988, since the sum level of all such subsystems must be not less than 0.988.

Therefore, each subsystem has been evaluated to determine that the basic subsystem achieves the minimum reliability level and what changes may be necessary to efficiently augment subsystem reliability.

4.2.2 Subsystem Reliability Evaluation

The results of the subsystems analyses are summarized in table 4.2-1. The reliability values shown are based upon the diagrams shown in figures 4.2-1 to 4.2-5. Each subsystem has been evaluated for mission durations from 14 to 45 days (one case of 135 day resupply flight) for both C and D configurations. Details of each

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configuration and/or modification will be discussed on a subsystem basis to emphasize the approach. No major problem areas have been identified, although the ECS appears to represent an area requiring more detailed study.

Fuel Cells

Reliability estimates were obtained from the Apollo-X Study (Contract NAS9-3140) report no. SID-64-1860 vol 7, and applied, assuming an exponential distribution (pure generic failure rate) for a fuel cell assembly. The number of assemblies necessary to provide power for each flight was identified and then evaluated for reliability. Reliability improvement may be obtained by providing redundant fuel cell assemblies in either hot or cold standby. The cold standby condition offers the greatest reliability increase and eliminates any dependency on premature cell wearout. Precautions must be taken to ensure compliance with crew safety conditions, and to assume the availability of sufficient power to activate the standby cell, in the event of a primary power failure in one of the operating cells. There are particular advantages associated with sharing spare cells with the CSM. These are discussed in the section of LEM/CSM backup potential. Figure 4.2-1 illustrates a typical fuel cell configuration and table 4.2-1 tabulates the reliability levels associated with each flight configuration. In general, cold standby has been selected for reliability enhancement, since this study is permitted to assume inflight startup capability.

Cryogenic Storage

Examination of the CSM Block III cryogenic storage subsystem for the LEM Lab missions has evolved the recommended configuration shown in figure 4.2-2. Variations from the typical system detailed in report SID-64-105 vol. 9 (Apollo-X Study) include the recommended use of quad check valves in the tank lines, redundant pressure switches, pressure transducers and pressure relief valves. The primary reasons these variations are to

- enhance the probability of mission success
- eliminate requirements for manual control of tank motors and heaters which is not desirable for extended missions
- provide redundant pressure readouts at the pressure control points.

Squib valves in the tank lines are retained for CSM crew safety compatibility, although not required for the LEM Lab. Squib pressure relief valves are retained and are particularly useful in the LEM Lab, providing isolation of a failed relief valve.

Studies are being conducted to eliminate the need for vent valves (not shown), due to reliability considerations as well as CSM compatibility.

The following table summarizes the configuration recommendations and the justification for the minor variations.

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ITEM	RECOMMENDED CONFIGURATION	REASONS
Pressure Switch	Redundant switch with manual switch, to switch electrical control sig's. to heaters and motors.	• Eliminates need for manual, which would not be desirable for long duration flights. • Enhances mission success. • Easily modified for control
Pressure Transducer	Redundant (active)	• Serves as backup indication to isolate pressure switch failure and pressure transducer failure. • Enhances mission success • Easily modified
Pressure Relief Valve	Redundant (active) with squib backup	• Eliminates possibility of dependent failure which has redundancy or might only cause curtailment. Squib permits isolation. • Enhances mission success • Easily modified
Tank Line Check Valves	Quad check valves (Series - parallel redundant)	• Enhances mission success • Quad check is the only worthwhile method of enhancement • Easily modified

Reaction Control System

For flights requiring an RCS on the LEM Lab, every attempt has been made to use the present LEM RCS with a minimum impact on development and/or qualification. The RCS consists of equipment which is dependent on operating cycles and pressurized time. One method of minimizing the pressurized time on any component is to drain one fuel tank at a time, rather than to drain all fuel tanks simultaneously. In most cases, the resultant pressurized time will be less than, or equal to the present LEM pressurized time. In missions where the LEM Lab will be manned and separated from the CSM, crew safety must also be considered. Each of these flights will be evaluated separately, to define the abort requirements for docking with the CSM or supply vehicles. Generally, there are a number of alternative configurations which may be applied for redundancy of pressure regulators, etc. Each represent varying levels of efficient use of components and impact on the qualification and development programs. Figure 4.2-3 illustrates a typical RCS configuration. Final decision for an RCS configuration will depend on the variety of flight requirements (particularly Flight 523) and minimum projected developmental and qualification costs.

Environmental Control

Typical configurations for the LEM Lab ECS are shown in figures 4.2-4, and 4.2-5. Variations in the basic configurations generally involve redundancy of high failure rate items and redundancy applied to take advantage of existing LEM components (glycol pump package, etc.). Since detail block diagrams and failure mode breakdown of component failure rates are not readily determined from the multi-path model used for LEM, the reliability estimate for the LEM Lab in this study is particularly pessimistic. The total generic failure rate was assumed to cause component failure, thereby resulting in a worst-case analysis. Efforts are presently devoted to the review of failure effect analyses to determine a more accurate mission success model for the Lab ECS.

Other Subsystems

Among the remaining subsystems, further definition is required to perform detailed reliability evaluations. Some missions will require attitude reference sensors references supplied to CSM or LEM for stabilization control of the vehicle. The flight control subsystem may, therefore, include a reference system (star trackers, horizon scanners, earth trackers, etc.) Once the accuracy, control duration and flight control requirements are more specifically defined, a reliability estimate may be made.

For the purposes of reliability estimation, data handling for the LEM Lab has been considered as part of the experimental payload. Since ground communications is performed by the CSM, only data handling, intercom, and experimental communications equipment is supplied in the Lab. When detailed information is available for individual flights, a reliability estimate will be performed. Again, accurate estimates and configurations will evolve due to the maximum utilization of existing hardware.

4.2.3 Spare Equipment List

Spare equipments, for the purposes of summary, are assumed to include all added equipments as well as physically replaceable items. Spare equipment must be provided with discretion for long duration space missions because:

- Little is known at this time regarding the capability of performing in-flight maintenance in space.
- Critical time in the failed-state is extremely important and may dictate some form of backup regardless of spares.
- The sole objective of the earth-orbiting mission is experimental; time spent in maintenance detracts from the mission objectives.
- Variations in subsystems requirements from flight to flight will mean variations in time spent in maintenance.
- Suitable fault isolation equipment must be provided.
- The automatic mode of repair is far more desirable in most cases, due to short time of functional loss, and may easily be provided with manual check and override.

The spare equipment list presented in table 4.2-2 includes all items which have been added as redundant and as physical spares. This list is preliminary and will be subject to analytic refinement as well as review of the development and qualification program details. The subsystem reliability evaluation table (4.2-1) is based upon the incorporation of spares according to this schedule. Estimated operating times on specific subsystems must be accurately defined for each flight to completely define these requirements.

4.2.4 LEM/CSM Backup Potential

A brief study has been performed to obtain an estimate of the increase in mission success probabilities that could be obtained by the inclusion of subsystems in the Lab which back up those planned for use in the CSM. This study was performed because it was suspected that significant mission success improvements are achievable and further that the probabilities of abort early in the mission can be greatly reduced by operating the subsystem in a particular manner.

The results are shown in tables 4.2-3 and 4.2-4. Table 4.2-3 shows the estimated mission success probabilities for each subsystem. The first column of apportioned CSM numbers are taken from Report no. 64-105, vol 21 (Apollo-X Study, Contract NAS9-3140). The second column represents the estimated mission success probabilities for the corresponding LEM/Lab subsystem. The third column lists the total mission success probability if the Lab subsystem is not used to back up the CSM. The fourth column lists the new total probability assuming that the Lab subsystem is used to back up the CSM. For example, if the fuel cells in both the Lab and the CSM are operated in a certain sequence, the probability of an abort in the first 14 days can be reduced by a factor of 100.

In general, the major point is that success probability gains can be shown if the subsystem in the Lab is used in orbit before using the corresponding subsystem in the CSM. This is summarized in Table 4.2-4 where it is shown that abort probabilities can be reduced by nearly 100% in some subsystems.

Table 4.2-1 Subsystem Reliability Evaluation

Flight No. (See Note 10)	Mission Duration days	FCA (See Note 9)	Cryogenics	Battery	Pwr Dist	ECS (See Note 5)	RCS	Flt Cent (SCS)	Data Handlg. Comm.
209	14	N.A.	N.A.	.99980	1.000	.993736	N.A.	N.A.	
211-C D	30	.998821 .999995	.998832 .99830		.999980 .999980	.986225	N.A. N.A.	NOTE 8	
507	14	.999950	.999352		.99924	.994667	N.A.	NOTE 8	
509	"	.999950	.999098		.99924	"	.99470	.99300 ⁽¹⁾	
215	"	.999950	.999352		.99924	"	.998668	.996882 ⁽²⁾	
218-C D	45	.999454 .99858	.998330 .997556		.99975 .99975	.984136 "	.999180 ⁽⁴⁾ .999180 ⁽⁴⁾	.996382 ⁽⁴⁾ .996382 ⁽⁴⁾	
219-C D	"	.999454 .99858	.997943 .996782		.99975 .99975	"	N.A. .999180	NOTE 8	See Note 7
513	14	.999950	.999042	See Note 6	.99924	.994667	N.A.	NOTE 8	
221-E F	45	.999454 .99858	.998330 .997556		.99975 .99975	.984136 "	.99400 N.A.	Not Complete	
516-C D	"	.999454 .99842	.998330 .997943		.99975 .99975	"	N.A. Not Complete	NOTE 8	
518-C D	"	.99786 .99786	.998330 .997556		.99975 .99975	"	N.A. ⁽³⁾ .998668	N.A. .996882	
521-C D	"	.999454 .99842	.998064 .997160		.99975 .99975	"	N.A. N.A.	NOTE 8	
523-C D	" (135 Day Resupply)	.998454 .99858	.998330 .997943		.99975 .99975	Rel Data Provided For 45 Day Inter- vals-	Not Complete	Not Complete	
529-C D	"	.999454 .99858	.998330 .997556		.99975 .99975	"	N.A. Not Complete	N.A. N.A.	

Table 4.2-1 (cont.)

Flight No.	Mission Duration days	FCA (See Note 9)	Cryogenics	Battery	Pwr. Dist.	ECS	RCS	Flt Cont (SCS)	Data Hndlg. Comm.
230-C D	45	.999454 .99858	.998330 .997556		.99975 .99975	.984136	N.A. N.A.	N.A. N.A.	
DOD-1C 1D	30	.99980 .999458	.998832 .998832		.99998 .99998	.988225	N.A. N.A.	NOTE 8	
DOD-2C 2D	"	.999163 .999163	.998832 .998832	See Note 6	.99998 .99998	"	N.A. N.A.	N.A. N.A.	See Note 7
DOD-3C 3D	"	.99980 .999458	.998832 .998832		.99998 .99998	"	N.A. N.A.	N.A. N.A.	
DOD-4	14	.99995	.999352		.99924	.994667	N.A.	N.A.	
DOD-5	"	.99995	.999352		.99924	"	N.A.	NOTE 8	
DOD-6	14	N.A.	N.A.	(Not Complete)	1.000	.994667	N.A.	N.A.	
DOD-7a-C D	30	.996432 .99500	.99780 .99751		.99998 .99998	.988225 "	N.A. N.A.	N.A. N.A.	
DOD-7b-C D	"	.999540 .999382	.997242 .996710		.99998 .99998	"	N.A. N.A.	N.A. N.A.	

- Notes:
- (1) Does not include communication antenna - only counted for 30 hours unmanned-is-the probability of commencing next flight.
 - (2) Does not include star tracker (1 needed) and horizon scanner (one in CSM also) and data processor.
 - (3) Based on estimated operating time for RCS and LEM-ATCA of 15 days (4051b propellant).
 - (4) Based on estimated operating time for RCS and SCS of 10 hours.
 - (5) Environmental Control reliability estimates are based on a preliminary evaluation representing a worst case. Total generic failure rates have been used for all components without regard to failure modes or alternate modes of operation. Each failure has been assumed to cause total loss of function.
 - (6) Estimates for peaking battery reliabilities will be generated when accurate sizing data becomes available. Weight data assumes no redundancy requirements for peaking batteries.
 - (7) Data handling has been assumed as part of experimental payload and is not included in LEM/LAB support subsystems. Accurate estimates may only be generated for each flight when data handling requirements are completely identified.
 - (8) Flight controls consist of LEM/LAB providing G&N reference data to the CSM via star tracker(s), horizon scanner and/or earth scanner. Reliability will be made when reference requirements are firmed (accuracy, etc.).
 - (9) Spare FCA'S have been considered in cold standby. In cases where only one FCA is required a battery or CSM power is assumed available for cold start up.
 - (10) These values include spare parts sched. described in table 4.2-2.

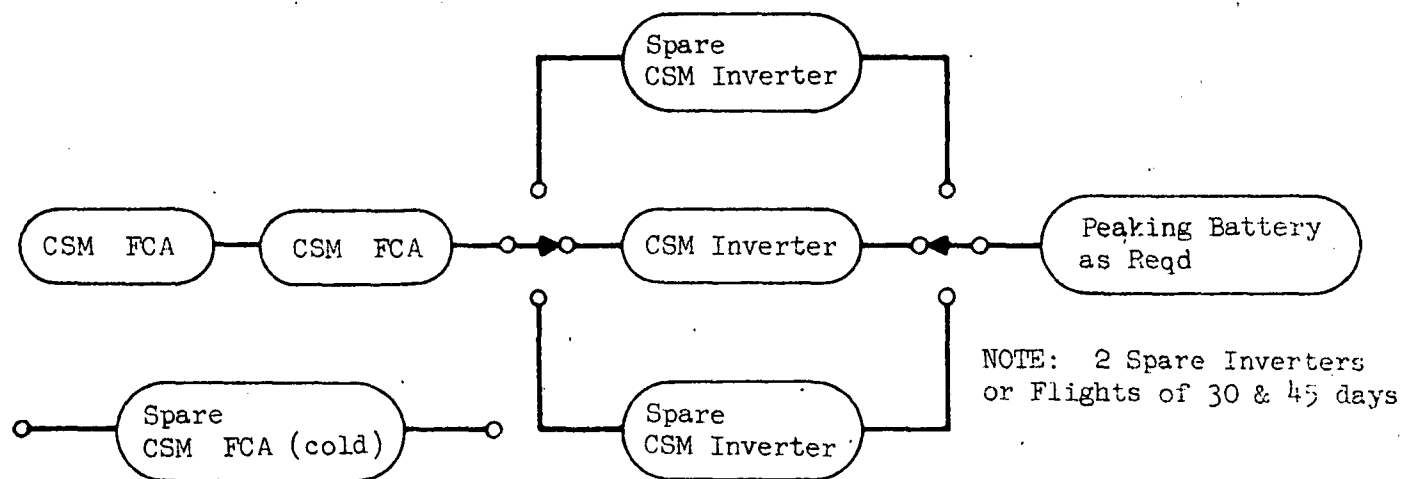
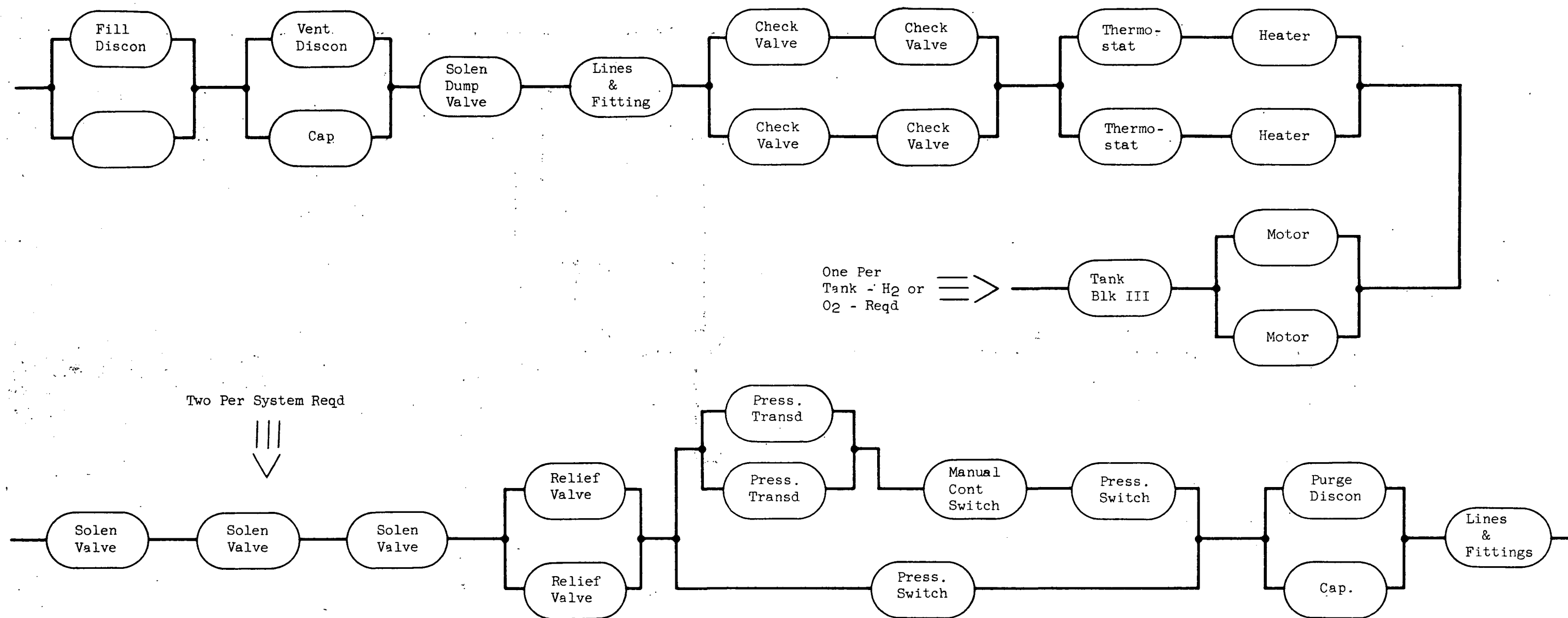


Fig. 4.2-1 Typical LEM Lab EPS Reliability Block Diagram
(Requiring 2 FCAs and 1 Inverter)



#110

Fig. 4.2-2 Typical LEM Lab Cryogenics Reliability Block Diagram

TOTAL REQUIREMENT DEPENDENT ON PARTICULAR FLIGHT

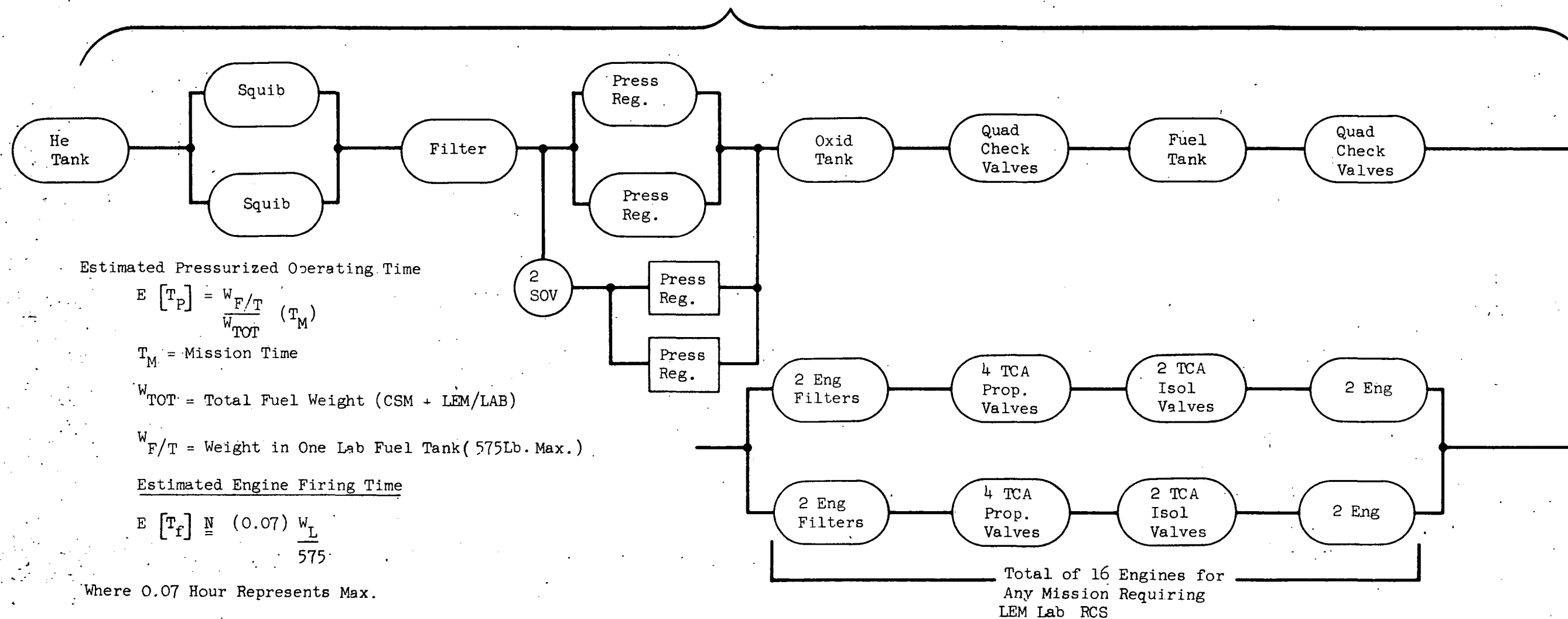
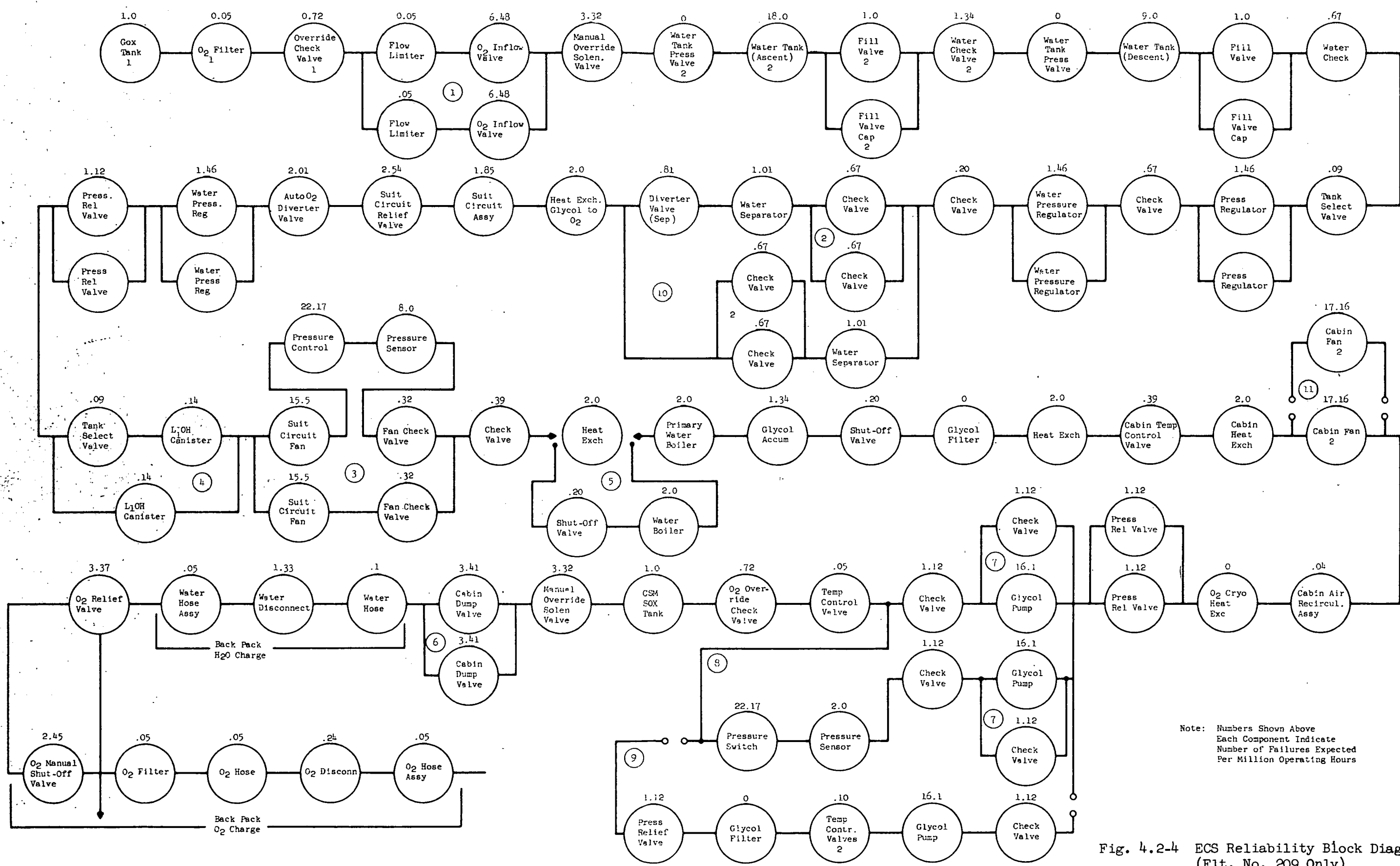


Fig. 4.2-3 Typical LEM Lab RCS
Reliability Block Diagram



Note: Numbers Shown Above
Each Component Indicate
Number of Failures Expected
Per Million Operating Hours

Fig. 4.2-4 ECS Reliability Block Diagram
(Flt. No. 209 Only)

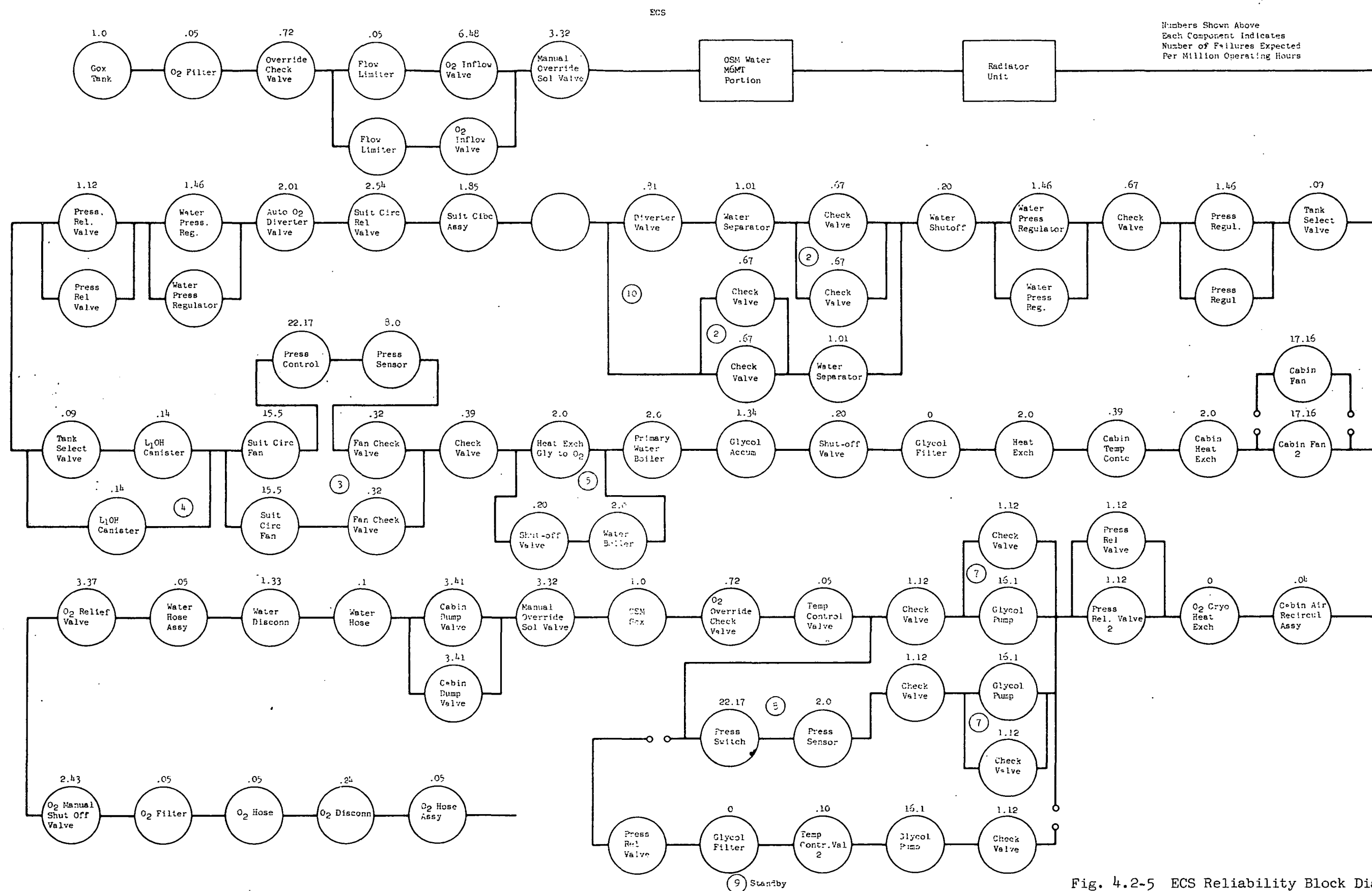


Fig. 4.2-5 ECS Reliability Block Diagram
(Except Ft. No. 209)

TABLE 4.2-2. SPARE EQUIPMENT LIST

Subsys	Item	No. Spare Items for (Days)				Remarks
		14	30	45	135	
EPS	FCA	1	1 2(DOD7b)	1	Re-supply	Spared Per Flight-Cold Standby Considered
	Inverters	1	2	2	"	Standby Redundant
	Cryo Relief Valve	2	2	2	"	Redundant (Squib For Shutdown) (One H ₂ , One O ₂)
	Cryo Press Xducer	2	2	2	"	Redundant (One H ₂ , One O ₂)
	Cryo Press Switch	2	2	2	"	Redundant With Manual Select (One H ₂ , One O ₂)
	Primary Batteries	2	N.A.	N.A.	N.A.	2 CEM Ascent Batt's. As Spares For Flight 209
ECS	Coolant Loop Water Boilers	Refer	to	Remarks		Existing LEM Boilers Qualified For LEM Mission (200lb H ₂ O)-Carried In Accordance With H ₂ O Handled.
	FCA Water Boilers	"	"	"	"	
	H ₂ O Press. Regulators	2	4	4	6	Redundant (1/Regulator)
	Pressure Regulators	1	2	2	4	Redundant
	Pressure Relief Valve	2	4	4	6	Redundant (1/Relief Valve)
	Cabin Fans	2	4	4	6	Spare (2 Fans/Assy)
N & G	Star Trackers	0	0	1	Requires Detailed Definition (Flt. 523)	Redundant (Estimated)
	Horizon Scanner	0	0	1		
	Earth Tracker	0	0	1		
RCS.	None					Subject To Refinement-(Flt. 523) Existing LEM RCS Considered

Table 4.2-3
MISSION SUCCESS PROBABILITIES

Subsystem	Apportioned CSM Mission Success	Estimated LEM/LAB Mission Success	Total Mission Success	Mission Success using LEM/LAB as backup	Remarks
EPS Conf D	0.999454 (14.8 days)	0.99858 (30.2 days)	0.998035	0.999884	Scheme involves sharing spare FCA's between CSM and shelter. Also note that potential abort is pushed back in mission time. For example, "D" config. probability of abort within 14 days is reduced by 1/100.
Conf C	0.97900 (45 days)	0.999454 (45 days)	0.97846	0.99999	
ECS	0.9890	0.9950	0.9854	0.99484	Use CSM for 14.8 days - use LEM/LAB for remainder or until failed.
				0.9999858	Use LEM/LAB in orbit for entire mission or until failed, then CSM system. CSM may periodically operate during certain flights.
RCS	0.9987 (SM-RCS)	0.992 (Not used unless as backup or special flights)	0.9987	0.9999976	Use LEM/LAB RCS in orbit - SM Engines considered in standby, (remainder of system pressurized) use SM at end or if LEM/LAB fails.

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Table 4.2-3 (cont)
MISSION SUCCESS PROBABILITIES

Subsystem	Apportioned CSM Mission Success	Estimated LEM/LAB Mission Success	Total Mission Success	Mission Success using LEM/LAB as backup	Remarks
SCS	0.99388 (45 days)	0.9995 (Not used unless as backup)	0.99388	0.9999694	Use LEM/LAB in orbit for mission or until failed.
Propulsion	0.99994	0.9995 (RCS stored until end of mission when not used)	0.99994	0.9999698 0.9999817 0.99999357	Assume 50% Rel. RCS for de-orbit Assume 70% Rel. RCS for de-orbit Assume 90% Rel. RCS for de-orbit
G&N	0.990 (Experiment G&N)	0.98036 (PNGS not used if LAB not backup)	0.990	0.9998	Consider experiment G&N backup to LEM/LAB PNGS.

TABLE 4.2-4
SUMMARY OF POTENTIAL MISSION SUCCESS IMPROVEMENTS

Subsystem	Present aborts per 1000 Missions (CSM + Lab)	Revised aborts per 1000 Missions (CSM + Lab backup)	% Reduction in aborts per 1000 Missions with Lab backup
EPS	1.90 (D Config)	0.116	94
	21.80	Negligible	100
ECS	11.40	0.0142	100
RCS	1.30	0.0024	100
SCS	6.12	0.0306	100
Propulsion	0.060	0.00643	89
G&N	10.00	0.20 (Not abort item-used for experiment)	98

4.3 STRUCTURAL ANALYSIS

4.3.1 Introduction

These data describe structural design criteria, the structure, comparative design loads at the support trunnion points for LEM and LEM Lab, an investigation of a 30-inch deep rack and an examination of some of the equipment supporting structure.

Results of the rack loads comparison with the descent structure of LEM show that the 64-inch descent structure will require no major structural modification for application to the LEM Lab missions. The 30-inch rack will incorporate the necessary redesign changes compatible with load and stiffness requirements. Equipment supports are integrated into existing structure with the least possible amount of modification.

4.3.2 Structural Design Criteria

The structural design criteria to be used for the launch and boost phases of the LEM Lab Mission are basically the same as those being applied to the LEM design. These criteria are summarized below.

Limit Load Factors - Launch and Boost

- | | |
|--|--|
| a. End Boost Condition | $n_x = 4.9g$ n_y or $n_z = \pm 0.1g$ |
| b. Lift-off Condition | $n_x = 1.6g$ n_y or $n_z = \pm 0.65g$ |
| c. Max q Condition | $n_x = 2.07g$ n_y or $n_z = \pm 0.30g$ |
| d. End 2nd Stage Boost
(Engine hard over) | $n_x = 2.15g$ n_y or $n_z = \pm 0.40g$ |
| e. 1st Stage Cut-off | $n_x = -1.7g$ n_y or $n_z = \pm 0.1g$ |

The vibration environment currently being applied to the LEM will also be used for the LEM Lab configuration. It is expected that the LEM Lab basic structure will be designed for steady acceleration conditions. This assumption is based on the results of analyses made for LEM. The expected vibration inputs to equipment on LEM Lab are as follows:

- a. Input to equipment Supports from exterior primary structures - launch and boost

<u>cps</u>		<u>cps</u>	
<u>Random</u>	<u>5 min/axis</u>	<u>Sinusoidal</u>	<u>3 octaves/min</u>
10-23	12 db/oct.	5-18.5	0.20" D.A.
23-80	.025 g^2 /cps	18.5-250	3.5 g
80-110	12 db/oct.	250-330	0.0011" D.A.
110-950	.075 g^2 /cps	380-2000	8.0g
950-1200	12 db/oct.		
1200-2000	.025 g^2 /cps		

- b. Input to equipment supports from Interior Primary structure launch and Boost

<u>cps</u> <u>Random</u>	<u>5 min/axis</u>	<u>cps</u> <u>Sinusoidal</u>	<u>3 octaves/min</u>
10-23	12 db/oct	5-16	0.20" D.A.
23-80	.025 g ² /cps	16-250	2.5 g
80-100	12 db/oct	250-420	0.0008" D.A.
100-1000	.060 g ² /cps	420-1000	7g
1000-1200	12 db/oct.	1000-2000	8g
1200-2000	.025 g ² /cps		

4.3.3 Description of Structure

To achieve mission objectives with maximum utilization of existing hardware, the basic structural components of the ascent and descent stages of the LEM are to be used wherever possible. Some secondary structural component changes have been made to provide integration of equipment and subsystems for a particular flight.

4.3.3.1 Ascent Stage

The ascent stage consists of three major structural components, i.e. the main crew compartment, the midsection, and the aft equipment bay. The main crew compartment is designed in the form of a frame-stiffened welded aluminum alloy shell structure 92 inches in diameter. The frames are spaced approximately 10 inches apart; fore and aft load carrying capability is provided by main longerons, stringer stiffeners and beam members. Extensive use of welding provides maximum pressurization reliability. The forward face provides the main bulkhead for supporting (1) the forward interstage support points, (2) the window structure, and (3) the forward hatch. It also functions as the shell pressure bulkhead. The mid-section is made up of two major weld assemblies; the upper assembly contains the machined deck, a docking tunnel, ring, and the two cylindrical segments of the chem-milled skin. The lower assembly consists of a machined deck and two lower segments of the chem-milled cylindrical skins. The two weld assemblies are joined mechanically on each side at the X 253 longeron. Skins are stiffened by 2-inch deep circumferential frames spaced approximately 10 inches apart. The forward and aft ends of the midsection skin deck assemblies are attached to two machined bulkheads at $\pm$ Z27. The integrally stiffened bulkheads are machined from 2219-T87 aluminum alloy plate. All external support structure required to carry pressure loads and provide equipment support is attached to the integrally machined flanges on the bulkheads. The upper hatch is located in the mid-section.

The forward interstage connection is located at the bottom of the front-face beams and attaches to the forward beam structure on the descent stage. The aft interstage structure is laterally trussed from the -Z27 bulkhead and attaches to the aft descent stage mainbeam.

The above structural components are identical with those of the LEM. In the area of the aft equipment bay, there has been revision of equipment and subsystem requirements for particular missions. These items are integrated into the basic structure through the use of supporting members which attach to existing hard points on the basic structure.

4.3.3.2 Descent Stage - Rack

The LEM descent stage structural assembly, without landing gear and descent propulsion system, will serve as the equipment rack and ascent stage support for most of the flights.

The descent stage primary structural assembly consists of four main beams approximately 64 inches deep, arranged in a cruciform configuration interconnected by diagonal members. The beams are made of aluminum alloy conventional skin and stringer construction; chem-milling is used extensively to reduce weight. All joints are made using standard mechanical fasteners. A quadripod tubular truss structure is used at each of the four ends of the cruciform beams to transfer loads between the LEM Lab and the Spacecraft LEM Adapter (SLA). The diagonal bays are used to house tanks and equipment and scientific experimental packages. The unmodified descent stage LEM structure will be used in all those configurations which will accommodate experimental packages within the existing SLA envelope. For approximately six missions the experiment integration requirements are such that the existing 64-inch deep rack cannot be used. For these conditions a revised 30-inch deep rack will be designed and qualified for the design criteria of the LEM Lab. Preliminary structural requirements and designs have been established for feasibility and weight estimation.

4.3.4 Comparison of Design Load on LEM and LEM Lab at the SLA Pickup Points

Since the design loads of the adapter pick up points result from superposition of static inertia, redundant, wind, and thermal loads, it was necessary to perform a load analysis for the LEM Lab, and compare it to the LEM. The elastic properties of the SLA shell structure were obtained through the use of a computer program for shell structures. These were combined with the influence coefficients of the LEM, to give the interaction forces on the LEM and SLA. The LEM loads summarized herein are for a total launch weight of 32000 lbs. Included in the conditions investigated are the inertia, air and thermal loads. Inertia forces shown for the LEM-Lab are for a total weight of 21,000 lbs. A comparison of applied loads to the LEM trunnion points for the LEM and the LEM Lab design conditions shows that the new loads are well within the design envelope of the original adapter and rack assembly. See Figure 4.3-1.

4.3.5 Parametric Investigation of Rack Depth

The basic 64-inch depth structure of the LEM descent stage will be utilized as a rack to support the LEM Lab in the SLA and to provide experiment and equipment support for the majority of the missions. Since some experiments could not be integrated into the LEM Lab with the 64-inch rack, a parametric study was made to determine a minimum depth of rack for the design weights expected for most of the missions. Results of the analysis are shown in Figure 4.3-2; the structural weight is plotted vs. depth of rack for various design weights. As a consequence of this study, a depth of 30 inches was selected since it shows the least weight for the majority of the mission design weights. The basic design of the 30-inch rack incorporates the same interstage members, structural configuration, and SLA attachment trunnion as the LEM ascent stage.

The variation in design weight assumed for the analysis were 6000, 8000, 10000, 12000, 16000, 20000, and 32000 lbs. The depth used were 64, 45, and 30 inches. The materials selected was 7075T-6 aluminum alloy.

4.3.6 Investigation of LEM Aft Equipment Bay Load Carrying Capability for LEM Lab Equipment Integration.

An investigation was made to determine the amount of equipment which could be supported from bulkhead - Z27 on the Ascent Stage of LEM. A structural configuration for equipment support was designed using existing hardpoints on - Z27. The criteria used to determine the allowable weight and center of gravity limitation for the package was the horizontal load capability at the upper support points. The location of the support points and the allowable loads are shown in Figure 4.3-3.

The permissible weight and c.g. location of all the equipment to be supported from the bulkhead is shown in Figure 4.3-4.

Axes Notation: (LEM)

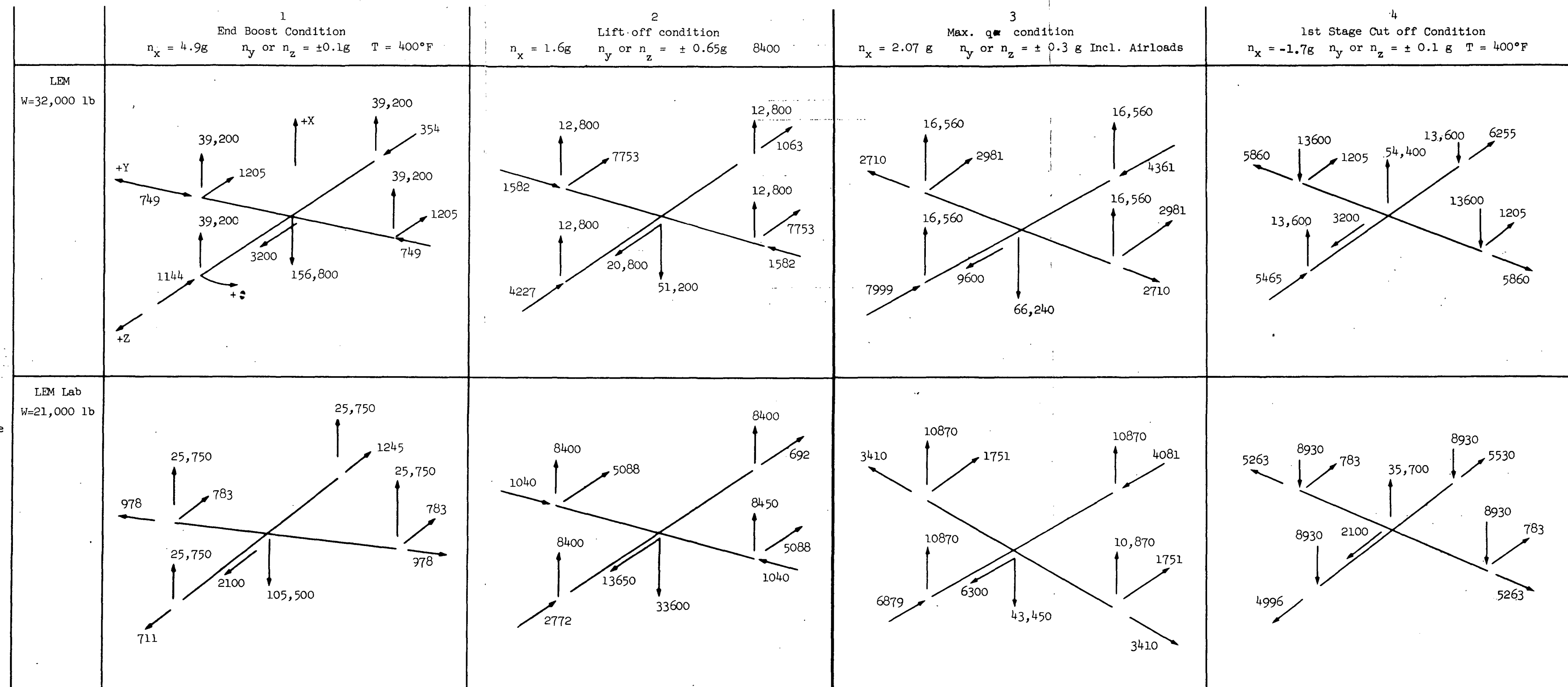
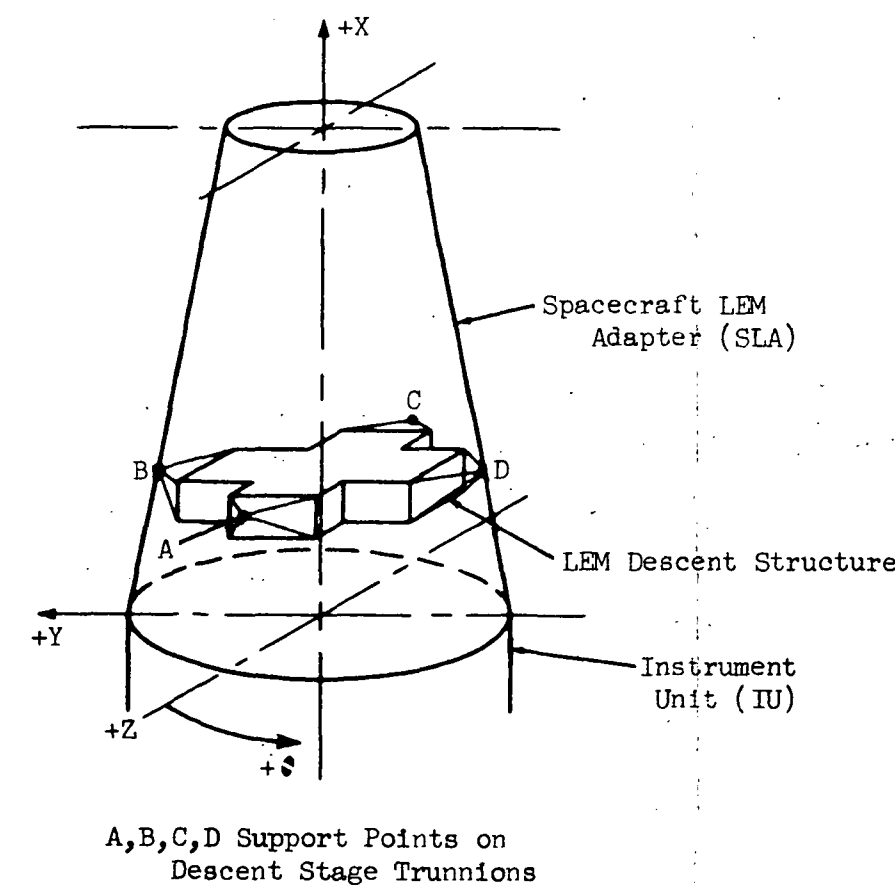


Figure 4.3-1. Total Limit Loads on LEM and LEM LAB (0°)

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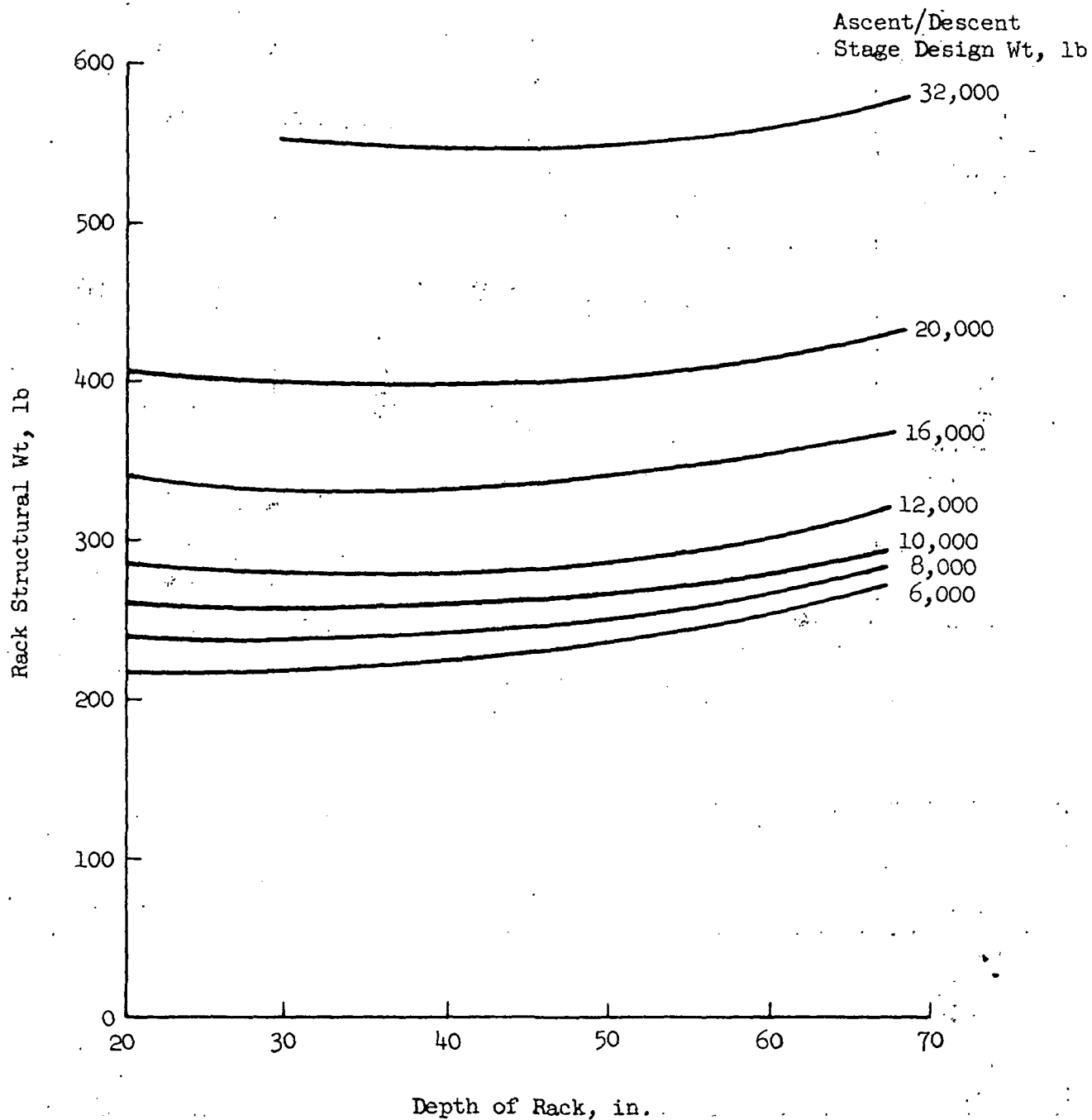
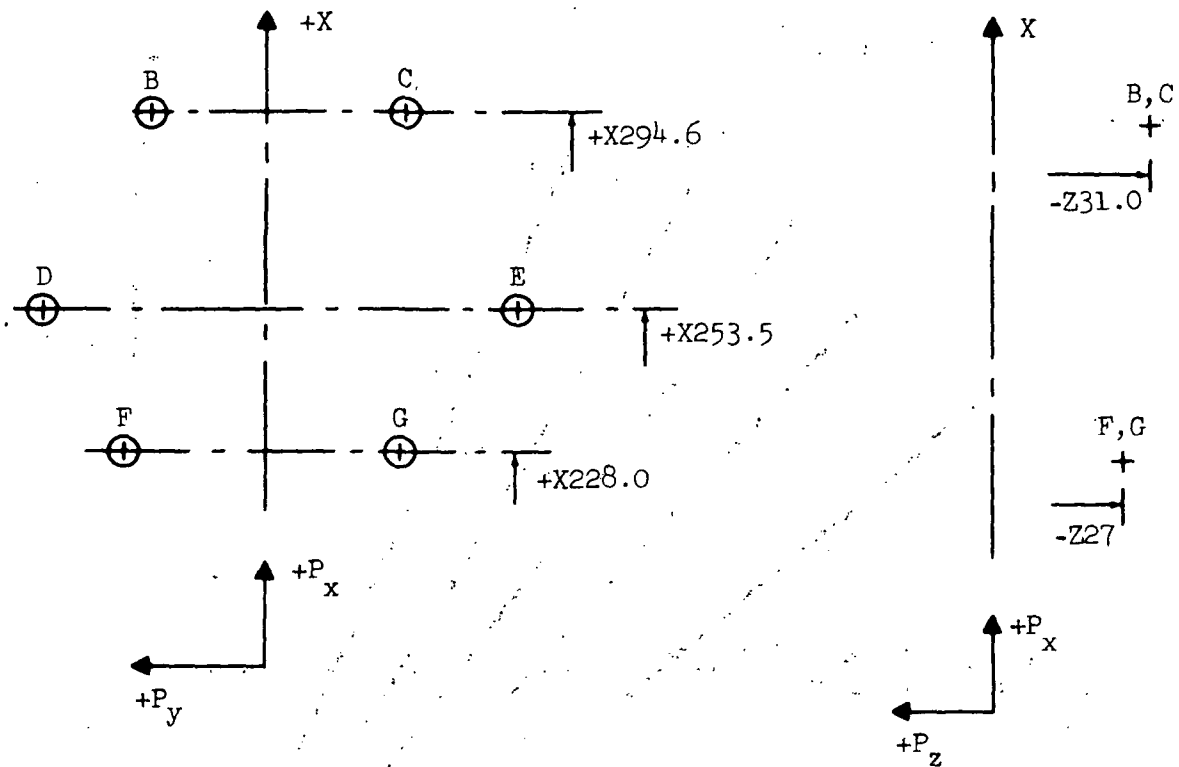


Fig. 4.3-2 Variation of Rack Weight vs. Rack Depth for Various Design Weights

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Support Points	Loads, lb		
	P _x	P _y	P _z
B	-1250	± 500	±1500
C	-1250	± 500	±1500
F	0	±6130	±2400
G	0	±6130	±2400
D	±7000	±6130	±1800
E	±7000	±6130	±1800

Note: Loads are limit applied to Spacecraft Structure.

Fig. 4.3-3 Allowable Loads on Bulkhead - Z27

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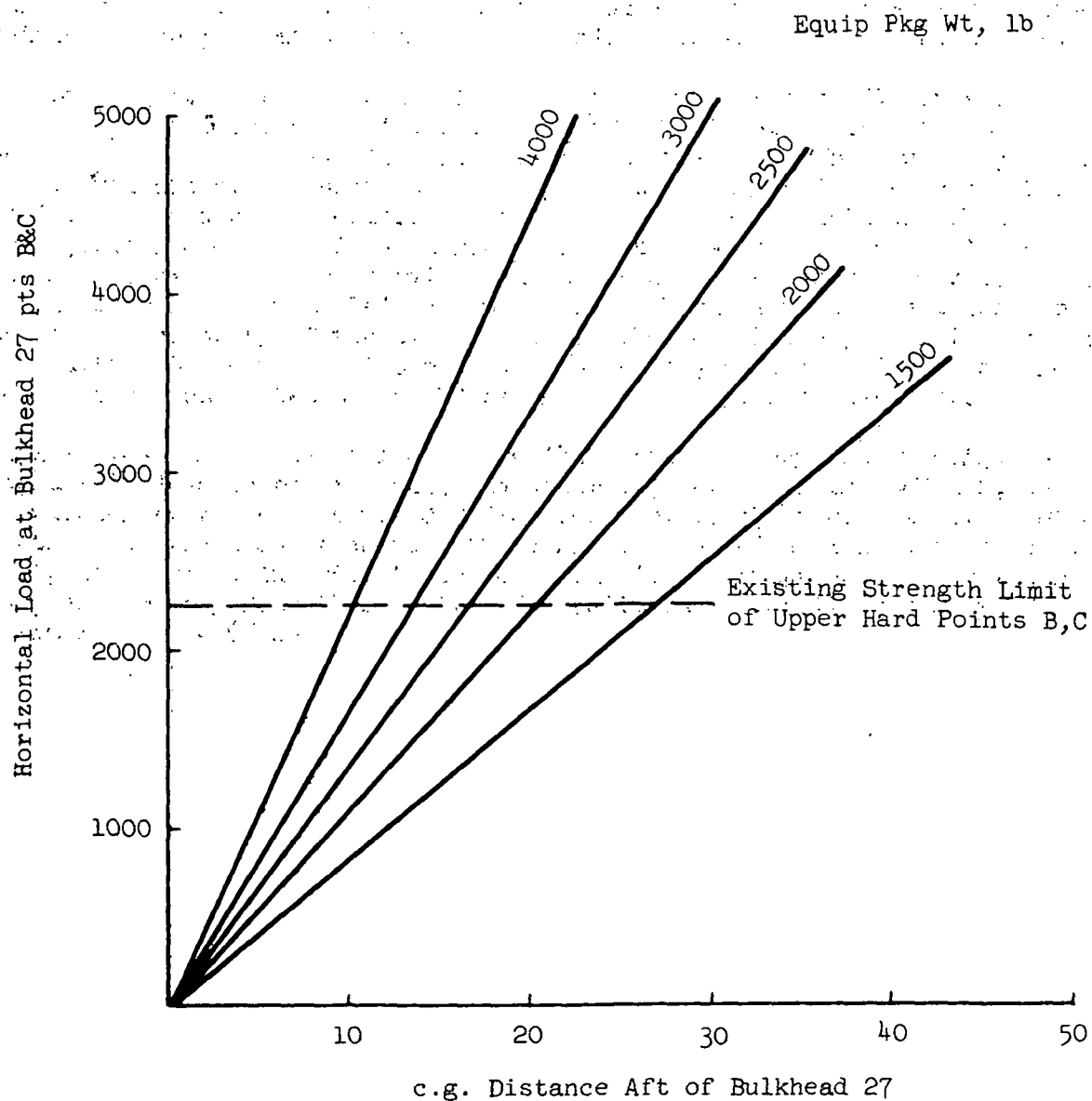


Fig. 4.3-4 Limitations of Aft Equipment Bay Loads

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4.4 PROPULSION OPERATIONS

The profile of propulsion operations for each mission is shown schematically in Figures 4.4-1 to 4.4-7. Propulsive velocity and coasting time data are presented for the SIVB and SM stages. Service Module main propellants are also indicated for the ascent and deboost phases, per NAA data, as shown in Table 4.6-3. Saturn IB is the booster vehicle for all low inclination, low altitude flights (200 series and DOD flights). Saturn V is the booster vehicle for Flight 523 (low orbit-low inclination - high payload), in addition to all polar and synchronous orbits.

The Synchronous orbit flights (Figure 4.4-2) utilize the SIVB in a dog-leg maneuver from 100 n mi circular orbit, comprised of 100 n mi-synchronous attitude transfer and a 14° plane change. At synchronous altitude the SIVB is used again in a combined burn maneuver consisting of a 14° plane change and a velocity increase. SIVB is jettisoned and the vehicle remains in a slightly elliptical orbit as the SM transposes and docks. At synchronous altitude (apogee of the elliptical orbit), the SM propulsion system provides the balance of the propulsive velocity required to circularize the vehicle's orbit. The synchronous flights utilize the full propellant capacity of the SIVB stage and the SM. These propulsive requirements, in addition to deboost (also orbital maneuvering on flight 509), require that the main propulsion system of the SM be utilized to its full capacity.

Figure 4.4-4 shows the orbital operations that are required in Flight 513: Following the 14 days in polar orbit, and after separation of the LEM Lab, the SM propulsion system boosts the CSM from 200 n mi to 670 n mi. At 670 n mi altitude, another powered phase transfers the vehicle to a 760 n mi apogee 670 n mi perigee elliptical orbit, in order to conduct Echo II observations. The re-entry maneuver is made directly from this orbit. The additional propellant required for these operations is 4265 lbs.

Flt No.	Inclin, deg	Orbital Wt, lb	Orbit Oper Time, days	Coast Time 80n.mi Park Orbit min	SPS Orbit Corr	CSM Wt, lb	SPS Propellant Data Supplied by NAA	
							Ascent	Retro
209	28.5	29,969	14	30	-	19,576	1,410	1,180
211C	28.5	31,102	30	30	-	20,833	1,410	1,349
D		30,776				19,447		1,191
218C	28.5	30,519	45	30	-	20,833	1,410	1,349
D		32,155				19,447		1,191
219C	28.5	36,512	45	120	x	21,766	1,410	1,349
D		36,497				19,447		1,191
221C	28.5	33,709	45	30	-	21,766	1,410	1,349
D		33,049				19,447		1,191
229C	28.5	34,039	45	120	x	21,766	1,410	1,349
D		34,358				19,447		1,191
230C	28.5	31,745	45	120	x	21,766	1,410	1,349
D		30,768				19,447		1,191
215C	51.5	30,732	14	225	-	19,576	1,345	1,180

Note: SPS Propellant Weight Includes 10% Reserve

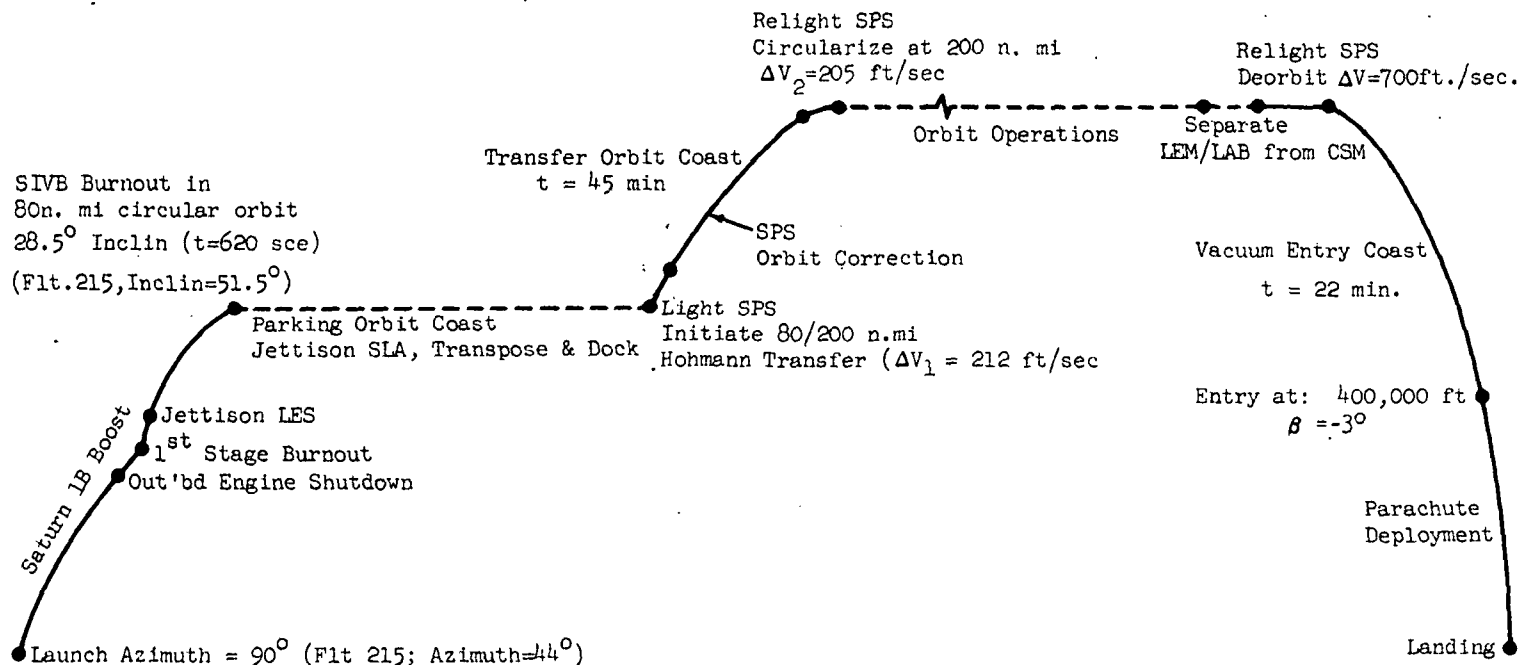


Fig. 4.4-1 Flight Profiles, Low Earth Inclination Orbits: 200 n.mi, 28.5 deg Inclination
(Flight 215: 51.5 deg Inclination)

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Flt No.	Orbital Wt, lb	Orbit Oper Time, days	SPS Ascent ΔV , ft/sec	CSM Wt, lb	Propellant Wt for Add'l Orbit Maneuvers, lb	SPS Propellant Data Supplied by NAA, lb	
						Ascent	Retro
509	50,567	14	3,530	19,614	Plane & alt changes: 2005	21,200	17,500
516C	59,877	45	3,050	22,818		21,200	21,500
D	55,189		3,280	19,485			17,500
521C	57,025	45	3,190	22,818		21,200	21,500
D	51,645		3,470	19,485			17,500

Note: SPS Propellants Include 10% Reserve

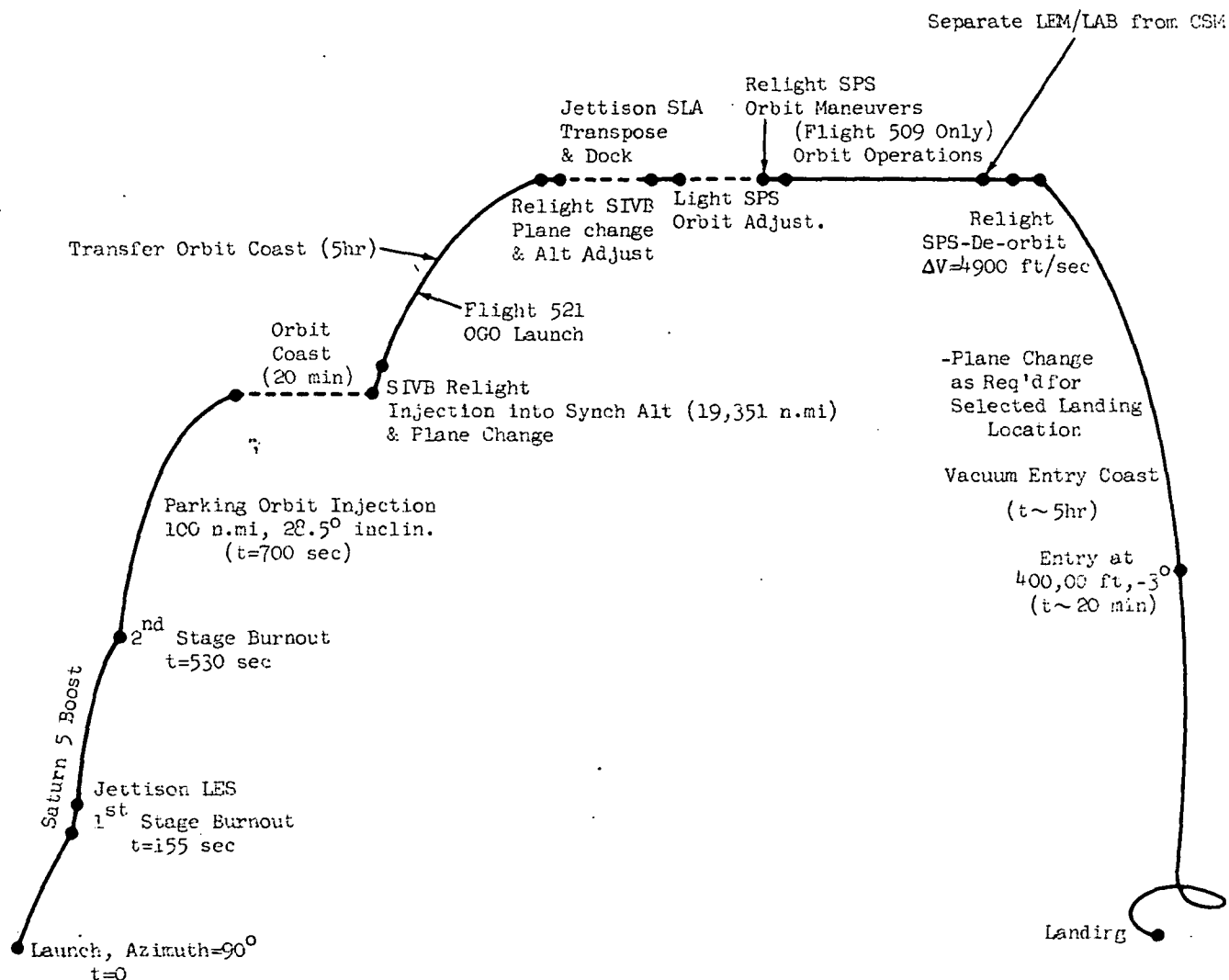


Fig. 4.4-2 Flight Profile, Synchronous Orbits

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4.4-4

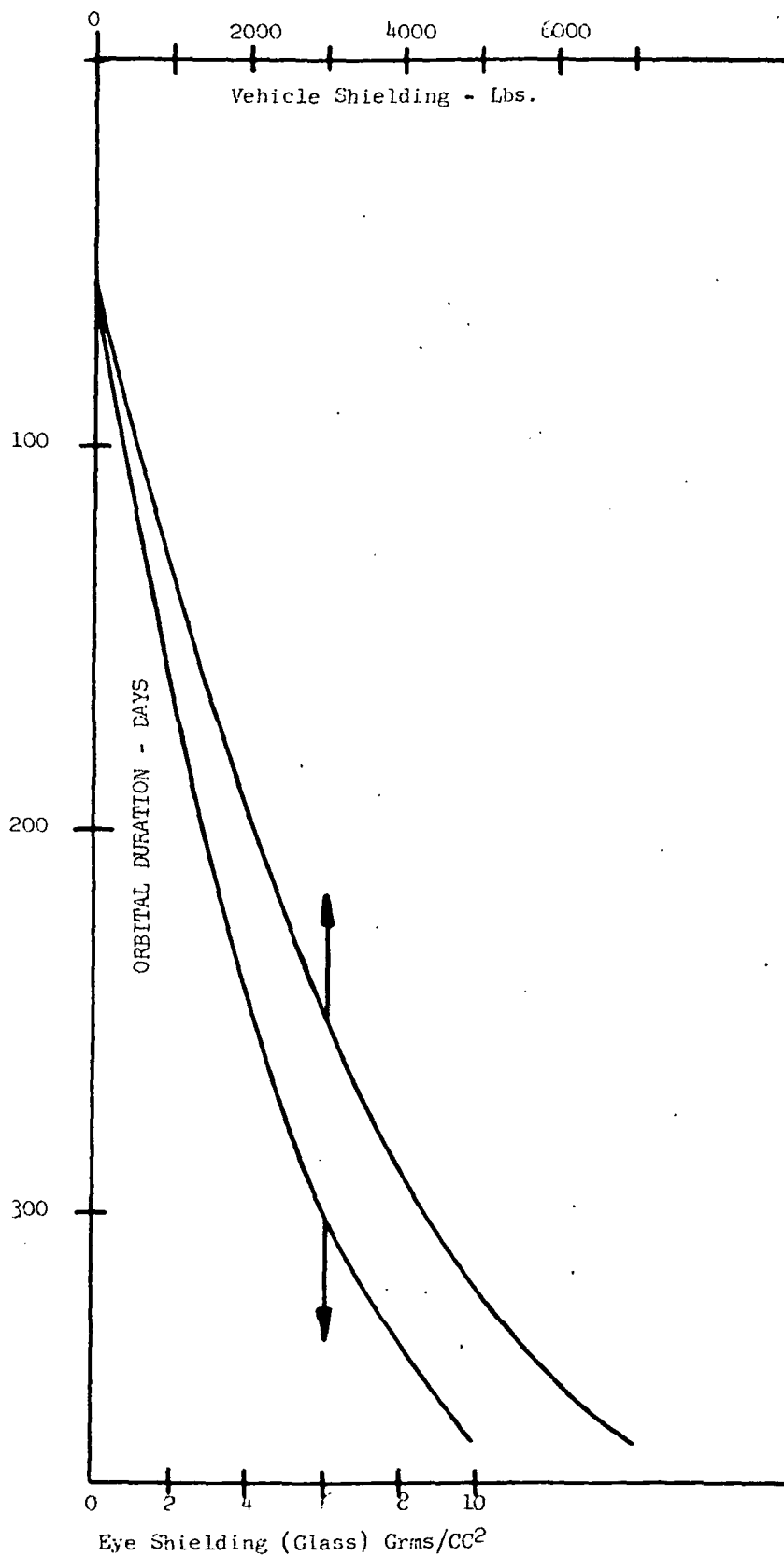


Fig 4.4-2 Radiation Protection; 200 n. mi Polar Orbit

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Flt No.	Inclin. deg	Orbital Wt, lb	Orbit Oper Time, days	CSM Weight, lb	SPS Propellant Data Supplied by NAA	
					Ascent	Retro
507	81.5	30,584	14	19,576	zero	1,180
518 C	97	38,159	45	22,812	zero	1,349
D		37,438		19,479		1,191
Note: SPS Propellants Include 10% Reserve						

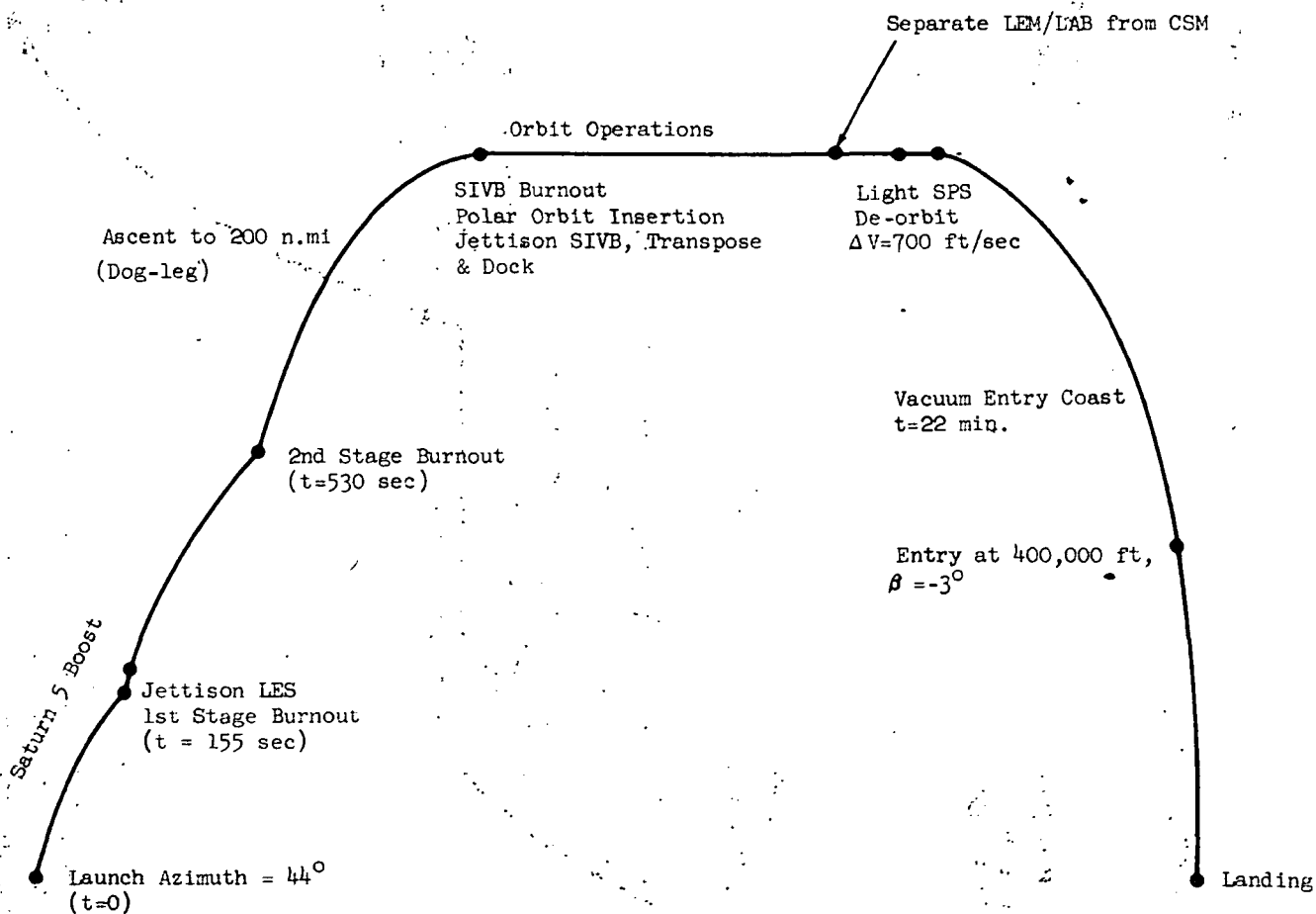


Fig. 4.4-3 Flight Profile, 200 n.mi; Polar Orbits

- Orbital Weight: 35,171 lb
- 200 n.mi, 81.5° Incln, 14 days
- CSM "Observation of Echo II": 670 n.mi perigee, 766 n.mi apogee, 81.5° Incln.
- CSM Weight: 19,608 lb
- SPS Propellants Include 10% Reserve.

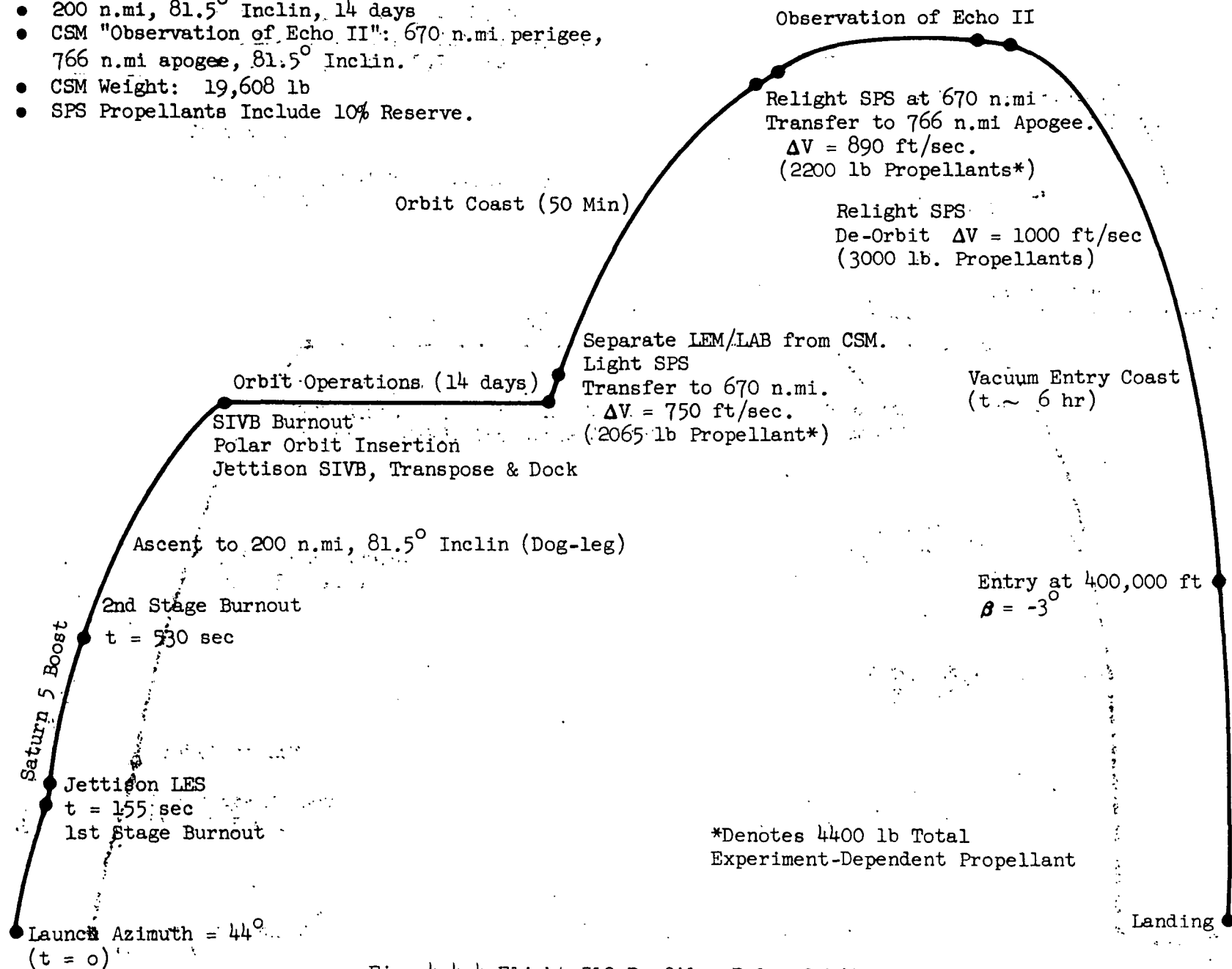


Fig. 4.4-4 Flight 513 Profile; Polar Orbits

Config. C

- Orbital Weight: - 44,193 lb
- 200 n.mi, 28.5° Inclination, 45 days.
- CSM Weight: 22,730 lb

Note: SPS Propellants
include 10% Reserve

Config. D

- Orbital Weight: 44,373 lb
- CSM Weight: 19,447 lb

Light SPS
De-orbit
 $\Delta V = 700$ ft/sec

(Config. C: 1349 lb Propellant)
(Config. D: 1191 lb Propellant)

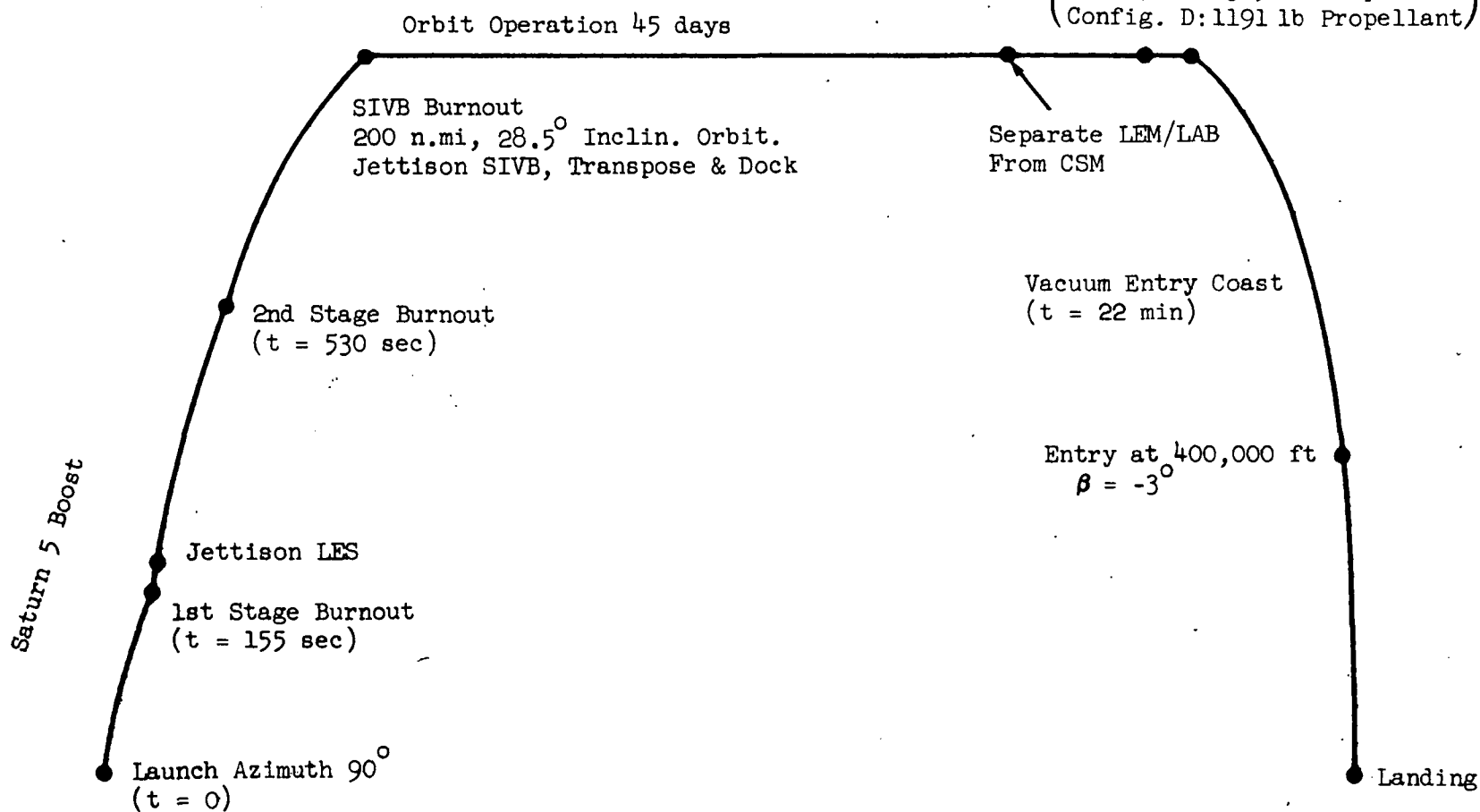


Fig. 4.4-5 Flight 523 Profile; 200 n.mi, 28.5 deg Inclination

Flt. No.	Inclin. deg	Orbital Wt, lb	Orbit Oper Time, days	CSM Wt, lb	SPS Propellant Data Supplied by NAA	
					Ascent	Retro
DOD-1C	32	31,971	30	20,827	1,410	1,349
D		31,390		19,441		1,191
DOD-2C	32	32,220	30	20,827	1,410	1,349
D		31,955		19,441		1,191
DOD-3C	32	32,438	30	20,827	1,410	1,349
D		31,868		19,441		1,191

Note: SPS Propellants Include 10% Reserve

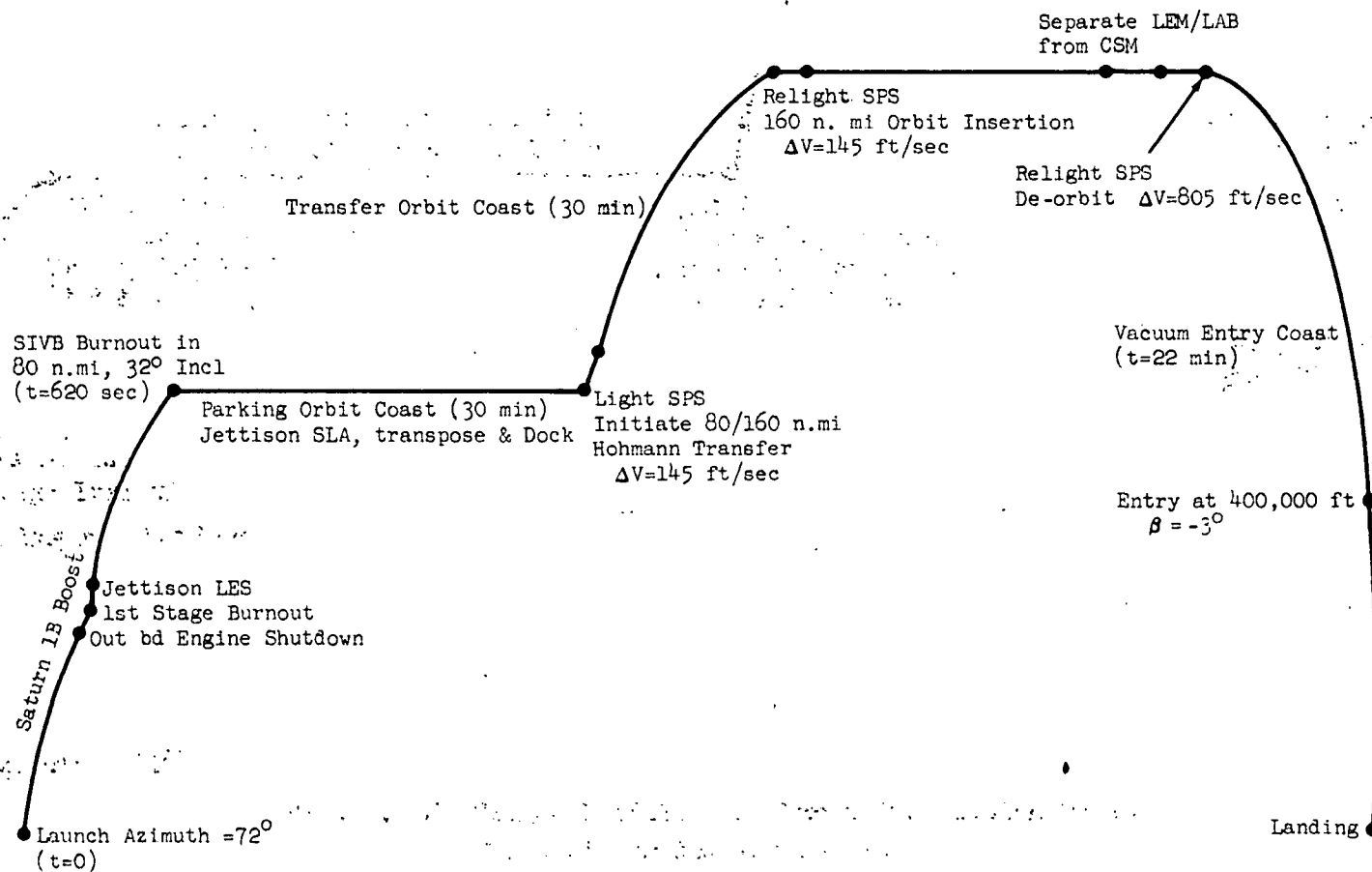


Fig. 4.4-6 Flight Profile; Low Inclination Earth Orbit; 160 n.mi, 32 deg Inclination

Flight No.	Inclin, deg	Orbital Wt, lb	CSM Wt, lb	SPS Propellant Data Supplied by NAA	
				Ascent	Retro
DOD-4	32	30,846	19,576	1,410	1,180
DOD-5	32	31,407	19,576	1,410	1,180
DOD-6	32	32,308	19,576	1,410	1,180
Note: SPS Propellants Include 10% Reserve.					

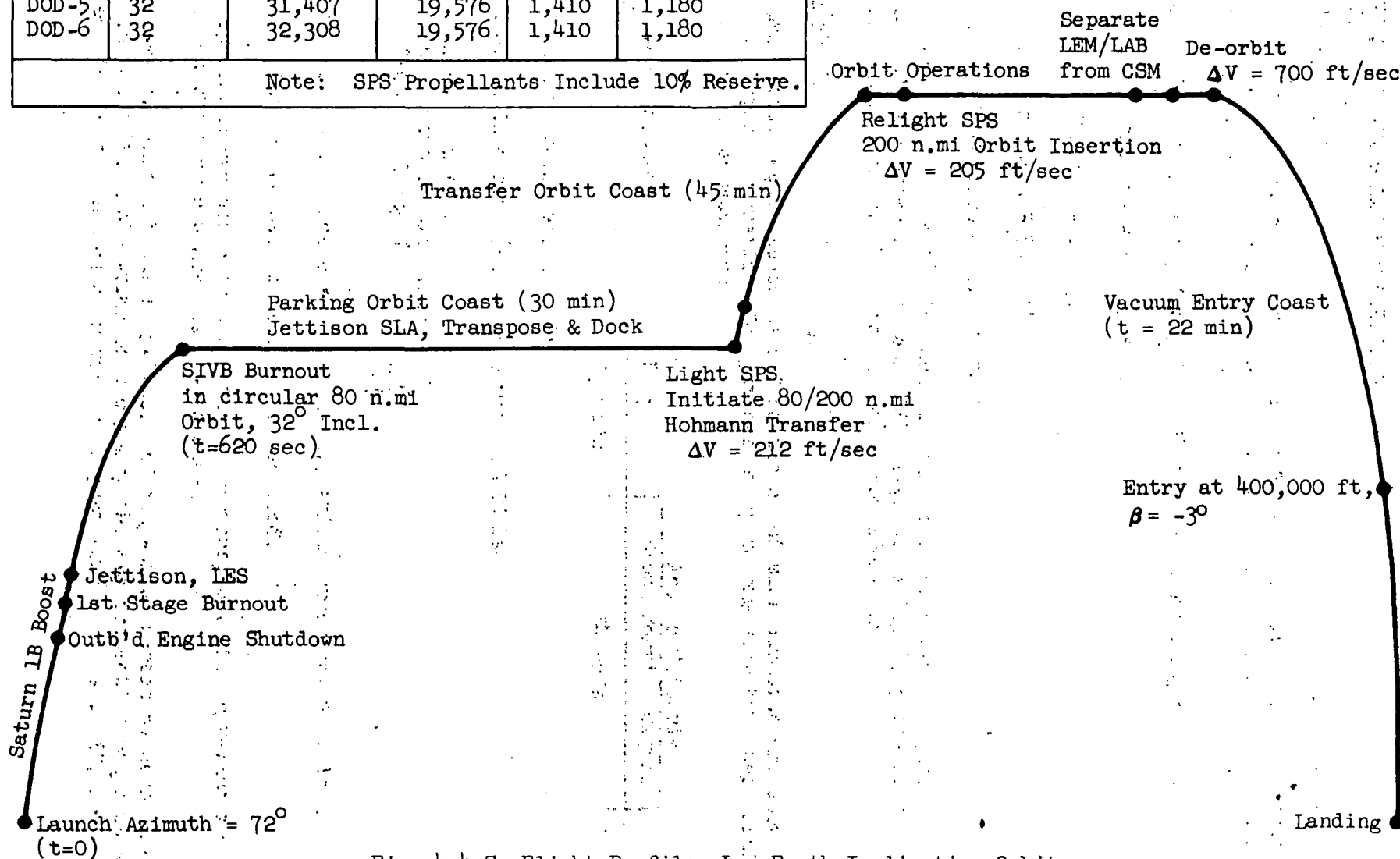


Fig. 4.4-7 Flight Profile, Low Earth Inclination Orbits;
200 n.mi, 32 deg Inclination

4.5 MICROMETEOROID/RADIATION PROTECTION

4.5.1 Micrometeoroids

Micrometeoroid protection requirements for a LEM Lab, as a function of mission duration and orbit, are shown in Figure 4.5-1. These data are based upon the following assumptions:

- Bumper Efficiency = 19% (weight = 19% of equivalent single skin)
- Meteoroid velocity = 30 km/sec
- Meteoroid Density = 0.5 gm/cc
- Lab critical area = 356 ft²
- Current LEM Meteoroid Flux Criteria

The results show the incremental increase in skin thickness (and weight) required over that already accounted for in basic LEM weights. A 0.9975 probability of no punctures has been used for the missions under consideration.

4.5-2 Radiation

Radiation protection requirements as a function of mission duration are shown in Figure 4.5-2. Exposure durations for periods of up to one year have been analyzed using the following assumptions:

• Allowable dosage:	Apollo Criteria		
	Eyes	Blood Forming Organs	Skin
	27 rads	54 rads	233 rads
• 200 naut mi polar earth orbit			
• Radiation Intensity (rads/year) within LEM lab			
	Bremsstrahlung	5	
	Electrons (Artificial)	100	
	Protons (Trapped)	80	
		185	

- Equivalent skin thickness of Lab is 1.5 gm/cm² spherical Al shield.

Using Apollo criteria, no skin shielding is required.

Blood forming organs are affected primarily by protons, but the criteria are satisfied since these organs are tissue-protected.

The eyes, however, limit orbital stay-time. If stay-time exceeds 54 days, the eyes must be protected. For a stay-time of one year, the eye protection required would be equivalent to about 4 inches of water or 10 grams/cm² of glass. If, for

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4.5-2

operational reasons, thick eyeglasses are prohibitive, the entire vehicle must be shielded, or the crew must be rotated within the 54 day period.

Weight of radiation shielding for the spacecraft as a function of mission duration is also shown in Figure 4.5-2. Should crew duration exceed 54 days, this weight would be added to the ascent stage's basic structure. This Figure illustrates the desirability of providing local eye shielding rather than shielding the entire ascent stage.

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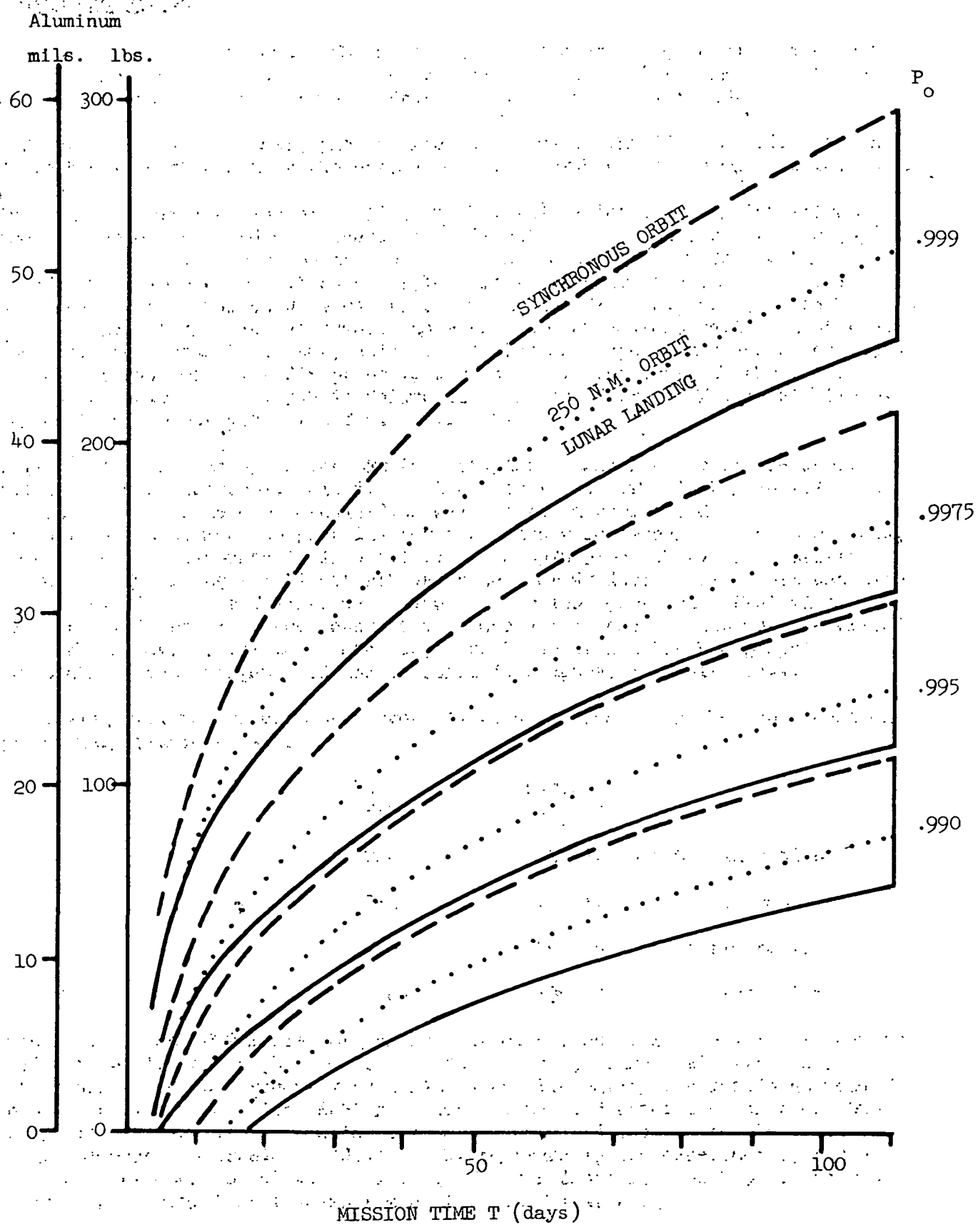


Fig. 4.5-1 Additional Micrometeoroid Shielding (Ascent Stage Dry Weight)

4.6 MASS PROPERTIES

Table 4.6-1 presents the basic LEM Lab ascent and descent stage structure weights used throughout this study. The table is based upon the 1 March LEM weight statement and describes all items removed from and added to the basic LEM.

A configuration weight summary showing the estimated weight of each flight is shown in Table 4.6-2 which compares the estimated and allowable weights and the remaining margin.

An integrated experiment equipment list (Table 2-1), a detailed LEM Lab weight statement (Table 3-4), and an integrated spacecraft weight statement (Table 3-5) appear with each flight mission and configuration description in Volume II. Command/Service Module propellant and allowable weights are tabulated in Table 4.6-3. These weight values were generated by NAA and Grumman, and approved by NASA for use in this study. The allowable laboratory and experiment payload must account not only for the total laboratory weight, but also CSM experiments and experiment-dependent expendables. The RCS propellant required is the total of the values appearing in Tables 3-4 and 3-5 in Volume II, plus the "Service Module RCS Propellant (Ascent and Re-entry)" value from Table 4.6-3.

Shielding values appearing in the LEM Lab detailed weight statements are derived by adding the micrometeoroid shielding weights from the curve in Figure 4.5-1 and the thermal and micrometeoroid shielding weights for a normal LEM. Additional shielding must be provided for the redesigned descent stage lower deck, and for the RCS system when it is utilized. A 0.9975 probability of no penetration was assumed. Shielding for Flight 218 was sized for 90 days. Flight 523 was sized for 135 days. All other flights were sized for nominal duration. Preliminary cg and moments of inertial of the integrated spacecraft in each flight were generated early in the study. These mass data were used to determine the RCS propellant requirements on each flight. CG and inertial data appear in Table 4.6-4.

The following flights are currently overweight, as indicated in the configuration weight summary (Table 4.6-2):

Flight	Excess Wt.
215	122 lbs
219	3659 lbs - D Configuration
221	293 lbs - D Configuration
229	5491 lbs - D Configuration
DOD-2	134 lbs - D Configuration

Flights 215, 221 and DOD-2 are marginally overweight, but relatively minor changes will effect sufficient weight reduction. For example, Flight 221 (Artificial Gravity Mission) can be brought within limits by eliminating the margin assumed for RCS propellants. In addition, NASA Headquarters has indicated that another 1000 lbs of payload capability is available from the Saturn I launch vehicle. Flight 219 is considerably overweight. However, by rearranging the experiments, the weight could be brought down to allowable limits. Since Flights 218 and 219 rendezvous, 219 carries RCS propellant for Experiment 1407 on Flight 218.

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If the RCS propellant for Experiment 1407 were deleted from Flight 219, and the 1500 Series of Experiments were transferred from 219 to 218, then Flight 219 (with the additional 1000 lbs of payload capability) would conform to weight requirements.

Flight 218 could not carry all the 1407 RCS propellant requirements and the 1500 Series of Experiments. It would, therefore, be necessary to transfer the 1407 Experiment, and the necessary RCS propellants to another flight with a large weight margin. Flight 518 has an allowable weight margin of 69,171 lbs, and sufficient available volume to accommodate the 1407 experiment.

Flight 229 is also considerably overweight. Again, experiment rearrangement could be accomplished by moving Experiments 1504 and 1302 from Flight 229 to Flight 523. Experiment 1306 could be moved from Flight 229 to Flight 230. All RCS requirements for the combined 229, 230 and 523 Flight could be carried aboard Flight 523. This would put Flight 229 within the allowable weight limit without imposing overweight on Flights 230 or 523.

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4.6-3

Table 4.6-1

Weight Breakdown of Basic LEM Lab Structure
(Based on LEM Mass Property Report LED-490-18 1 Mar 1965)

Subsystem/Remarks	Ascent	Descent
Structure Weight at Earth Launch	1,154.8	1,228.1
Remove: Thermal & M/M Shielding	- 224.3	- 118.3
RCS Thruster Supports	14.8	
RCS Tank Mounts	37.7	
He, O ₂ & H ₂ Tank Mts	- 10.0	- 33.8
Fuel Cell Mounts	- 23.9	
Oxidizer & Fuel Tank Mounts	- 14.9	- 110.4
Water Tank Supports	- 4.0	- 13.0
Equipment Supports	- 28.9	
Engine Cover/Structure	- 13.1	- 13.4
Handholds	- 8.0	
Scientific Equipment Supts		- 18.4
Lower Decks		- 56.4
Protective Coating		- 12.0
Steps/Platform		- 21.5
Separation System		- 29.0
Base Heat Shield		- 141.7
Add: Misc Supports Not Removable	45.0	10.0
Redesigned Lower Deck		65.4
Bolts Replacing Separation Syst		14.0
Basic LEM/Lab Structure	802.2	749.6
TOTAL	1,551.4	

Table 4.6-2

CONFIGURATION WEIGHT SUMMARY

Flt No	1			C			D		
	Est	Allow	Margin	Est	Allow	Margin	Est	Allow	Margin
209	9669	11,914	2245						
211				8881	10,488	1607	10,100	12,032	1932
507	10,182	85,707	75,525						
509	13,584	20,136	6522						
215	10,509	10,387	-122						
513	12,564	83,889	71,325						
218				8514	9555	1041	11,449	12,032	583
219				13,173	9555	-3618	15,691	12,032	-3659
221				10,507	9555	-952	12,325	12,032	-293
516				14,948	12,932	-2016	17,913	20,265	2352
518				13,974	82,334	68,360	16,654	85,825	69,171
521				12,611	12,932	321	14,844	20,265	5421
523				21,492	195,921	174,429	25,312	199,362	174,050
229				13,271	9555	-3,716	17,523	12,032	-5,491
230				9895	9555	-340	10,561	12,032	1471
DOD 1				9803	10,603	800	10,766	12,147	1381
2				10,939	10,603	-336	12,281	12,147	-134
3				10,072	10,603	531	11,157	12,147	990
4	10,127	11,579	1452						
5	10,689	11,579	890						
6	11,082	11,579	497						

NOTE: Values Shown Are For "Total Laboratory
And CSM Experiment Dependent Weights".

TABULATION OF CSM AND ALLOWABLE WEIGHTS

Applicable Flights	CONFIGURATION 1						CONFIGURATION C						CONFIGURATION D				
	209	507	509	215	513	DOD-4 DOD-5 DOD-6	211	221 218 219 229 230	516 521	518	523	DOD-1 DOD-2 DOD-3	211 218 221 219 229 230	516 521	518	523	DOD-1 DOD-2 DOD-3
Mission Duration - Days	14	14	14	14	14	14	30	45	45	45	45	30	30-45	45	45	45	30
Orbital Altitude - Nautical Miles	200	200	SYNCH	200	200	200	200	200	SYNCH	200	200	160	200	SYNCH	200	200	160
Inclination of Orbit - Degrees	28.5	POLAR	0°	50	81.5 Pos1	32	28.5	28.5	0	83 RETRO	28.5	32	28.5	0	83 RETRO	28.5	32
Command Module Weight - Pounds	(10,360)	(10,360)	(10,360)	(10,360)	(10,360)	(10,360)	(10,287)	(10,287)	(10,287)	(10,287)	(10,287)	(10,287)	(10,287)	(10,287)	(10,287)	(10,287)	(10,287)
Command Module Weight Empty	8946	8946	8946	8946	8946	8946	9018	9018	9018	9018	9018	9018	9018	9018	9018	9018	9018
Command Module Expendables	1414	1414	1414	1414	1414	1414	1269	1269	1269	1269	1269	1269	1269	1269	1269	1269	1269
Service Module Weight	(9216)	(9248)	(9254)	(9248)	(9248)	(9216)	(10,546)	(11,479)	(12,531)	(12,525)	(12,443)	(10,546)	(9160)	(9198)	(9192)	(9160)	(9160)
Service Module Weight Empty	8489	8489	8489	8489	8489	8489	8922	8922	9936	9936	9936	8922	8433	8433	8433	8433	8433
Service Module RCS Propellant (Ascent & Reentry)	74	106	112	106	106	74	74	74	112	106	74	74	74	112	106	74	74
Service Module Cryogenic Expendables	653	653	653	653	653	653	1550	2483	2483	2483	2483	1550	653	653	653	653	653
Service Module Retro-Propellant	(1130)	(1180)	(17,500)	(1180)	(3000)	(1180)	(1349)	(1349)	(21,500)	(1349)	(1349)	(1349)	(1191)	(17,500)	(1191)	(1191)	(1191)
Command-Service Module Subtotal	20,756	20,788	37,114	20,788	22,608	20,756	22,182	23,115	44,318	24,161	24,079	22,182	20,638	36,985	20,670	20,638	20,638
Allowable Laboratory & Experiments	11,914	85,707	20,136	10,387	83,889	11,579	10,488	9,555	12,932	82,334	195,921	10,603	12,032	20,265	85,825	199,362	12,147
Allowable Weight in Orbit	32,670	106,495	57,250	31,175	106,495	32,335	32,670	32,670	57,250	106,495	220,000	32,785	32,670	57,250	106,495	220,000	32,785
Service Module Ascent Propellant	1410	0	21,200	1345	0	1410	1410	1410	21,200	0	0	1410	1410	21,200	0	0	1410
SLA Adapter	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505	3505
Launch Vehicle Payload	37,585	110,000	81,955	36,025	110,000	37,250	37,585	37,585	81,955	110,000	223,505	37,700	37,585	81,955	110,000	223,505	37,700

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4.6-6

Table 4.6-4

Spacecraft cg and Inertia Estimates

Flight/Condition	cg's, in.			Inertias, slug-ft ²		
	x	y	z	I _x	I _y	I _z
209	31.0	1.7	- .7	15,370	101,560	103,238
211C	31.0	1.2	-1.0	20,239	127,982	125,799
211D	49.6	1.5	-1.6	19,030	124,391	123,215
507	46.6	1.6	-1.5	17,219	120,046	122,477
509	-24.6	1.5	- .6	27,316	181,079	184,503
215 No Extension	42.0	1.5	-1.3	17,212	119,817	122,255
100 ft Extension	407.3	1.5	-1.3	17,212	2,919,905	2,922,674
200 ft Extension	772.5	1.5	-1.3	17,212	9,748,320	9,751,089
513	21.8	3.2	-1.4	16,354	115,940	118,006
218C	21.4	1.0	- .4	18,118	120,435	121,779
218D	43.1	1.4	-1.3	17,723	123,493	125,593
219C	30.9	1.0	- .5	19,951	140,735	142,628
219D	51.6	1.4	- .9	19,282	140,325	142,929
218 Lab & 219 S/C C	109.3	.8	-2.3	24,072	369,613	372,887
218 Lab & 219 S/C D	142.8	1.1	-3.0	22,719	390,079	394,852
221 C No Extension	15.4	1.0	- .2	17,600	113,480	115,392
C 100 ft Extension	281.0	1.0	- .2	17,600	2,400,311	2,402,402
C 200 ft Extension	546.1	1.0	- .2	17,600	7,943,761	7,945,852
D No Extension	32.7	1.5	- .8	16,590	110,963	223,788
D 100 ft Extension	351.7	1.5	- .8	16,590	2,586,096	2,587,348
D 200 ft Extension	670.5	1.5	- .8	16,590	8,621,575	8,622,827
516C	-6.3	.8	- .3	34,263	250,711	258,222
516D	10.9	1.2	- .2	30,489	249,044	255,695

~~CONFIDENTIAL~~

Table 4.6-4

Spacecraft cg and Inertia Estimates (Cont)

Flight/Condition	cg's, in.			Inertias, slug-ft ²		
	x	y	z	I _x	I _y	I _z
518C	36.2	.1	.4	23,593	145,537	148,352
518D	62.9	1.2	-.6	21,446	149,376	152,688
521C	-13.0	.9	-.1	32,650	230,913	237,837
521D	6.5	1.2	-.1	29,285	241,554	244,209
523C	54.7	.5	-.6	25,485	179,071	183,414
523D	73.5	1.2	-1.3	24,545	163,431	168,546
229C	27.7	1.0	-.7	18,842	129,033	131,303
229D	48.4	1.4	-1.7	18,436	130,017	132,735
523 Lab & 229 S/C C	154.1	.7	-1.8	29,468	488,785	495,033
523 Lab & 229 S/C D	176.4	.9	-2.5	30,654	472,420	479,930
230C	4.4	1.0	2.0	18,688	84,200	84,683
230D	21.7	1.4	1.7	18,220	82,995	83,445
523 Lab & 230 S/C C	102.6	.7	0	29,372	290,267	294,670
523 Lab & 230 S/C D	120.3	.9	-.3	30,502	277,706	282,884
DOD-1 C	20.1	.7	-.4	19,331	126,036	126,339
DOD-1 D	39.0	1.4	-1.1	17,288	118,423	118,257
DOD-2 C	18.7	.7	-.3	19,626	124,429	124,437
DOD-2 D	39.6	1.4	-1.2	17,312	119,241	118,883
DOD-3 C	21.4	.7	-.4	18,940	127,266	128,900
DOD-3 D	40.3	1.4	-1.2	17,314	119,824	121,064
DOD-4	55.3	1.3	-.5	19,517	138,466	141,601
DOD-5	55.3	1.3	-.5	19,517	138,466	141,601
DOD-6	55.3	1.3	-.5	19,517	138,466	141,601

NOTE: xcg Measured Along Thrust Axis from Base of CM (Apollo Sta. 1000.0)
 ycg Measured from Thrust Axis
 zcg Measured from Thrust Axis

4.7 CSM - LAB INTERFACE

The proposed LEM - CSM docking umbilical for the Apollo missions consists of four #4 cables consisting of two runs of two cables each. These connections form the DC-DC interface for supplying power to the CM. Each line is sufficient in itself to carry the load but a backup cable set has been provided. It is anticipated that these cables will also carry the necessary wiring for the basic LEM status and condition monitoring. These cables, shown on Figure 4.7-1 are described as "Two #4 Cables".

An additional cable is required to carry the PCM TEA (instead of mode switch) output to the command module as is shown on the data handling schematic (Figure 3.4-1). This cable, called the "communication umbilical", consists of about eight coaxial cables and additional smaller wires.

The "100 psi oxygen line" discussed in the section on the environmental control system is also shown.

The interfacing requirements required by the flight control system will be accommodated either by the communication umbilical or within the bundles providing the DC-DC interface.

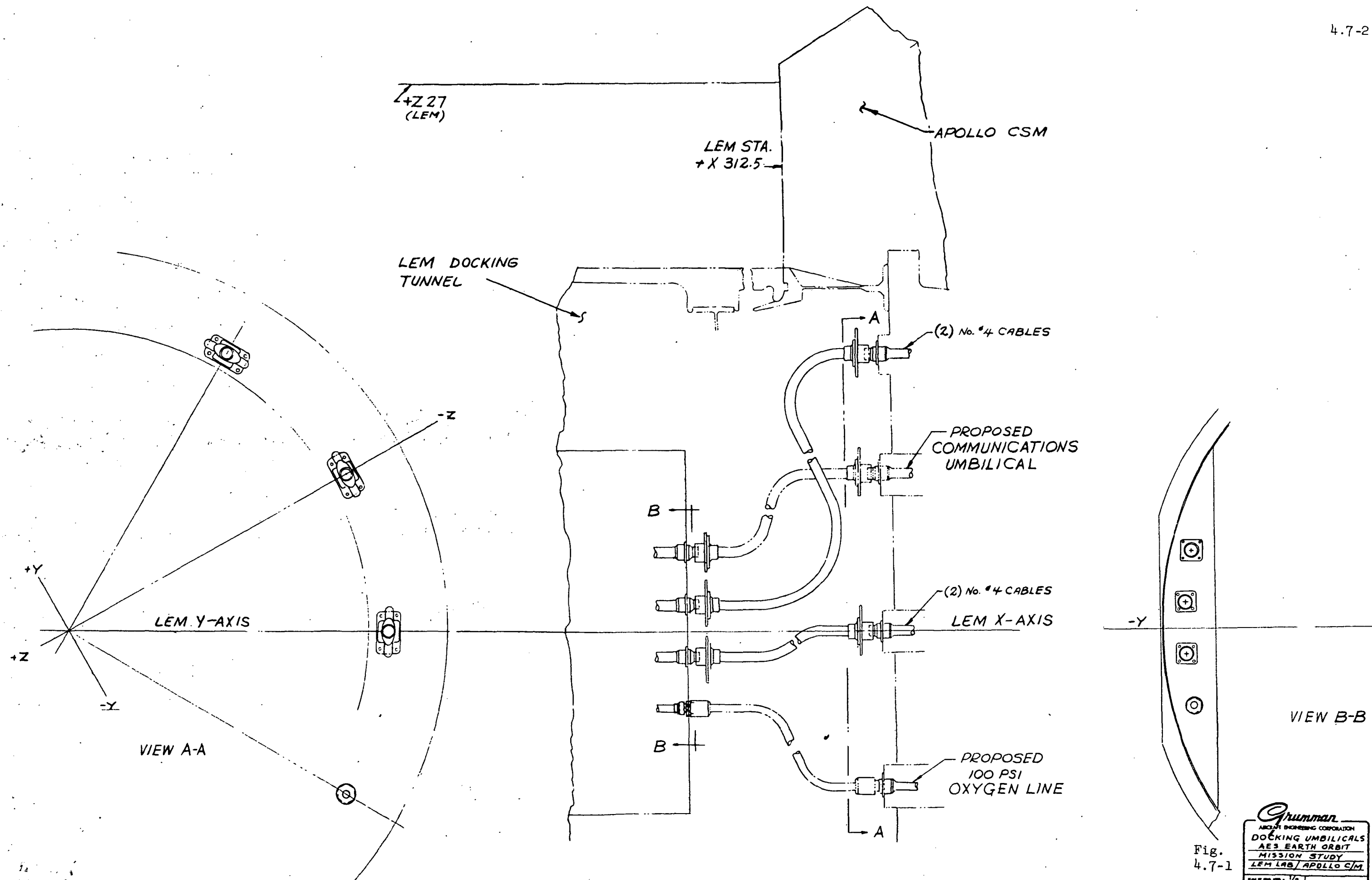


Fig. 4.7-1

Grunman
AIRCRAFT ENGINEERING CORPORATION
DOCKING UMBILICALS
AES EARTH ORBIT
MISSION STUDY
LEM LAB / APOLLO C/M
Scale of 1/2" = 1'-0"
AES 1011

5 DESIGN AND INTEGRATION

In addition to evolving integrated spacecraft configurations associated with NASA and DOD flights, design and integration efforts included the following activities:

- Establishing design/integration guidelines
- Identifying a reference LEM configuration
- Evolving standardized configuration modules
- Establishing experiment equipment and spacecraft subsystem commonality
- Identifying LEM structural modifications
- Performing volume utilization studies

Results of these studies are documented in subsequent sections.

5.1 GUIDELINES

- LEM hardware components should be retained in their original locations, where possible.
- No change is allowed to the LEM adapter.
- The full descent stage structure should be used where possible, but a reduced profile descent stage (30" instead of 68" depth) is permissible where large experiments cannot normally be fitted in the stack.
- The LEM/SLA pick-up structure should be preserved.
- Additional holes in the ascent stage pressure vessel are permitted only when demanded by the experiment.
- Design efforts should standardize locations of experiments, electrical power supply, and other Lab subsystems.
- A minimum of 7 days lithium hydroxide must be storable inside the lab.

5.2 REFERENCE LEM CONFIGURATION

Figures 5.2-1 thru 5.2-5 show the reference configuration of the LEM used for this study.

- Fig. 5.2-1 shows the structural arrangement of the LEM ascent stage
- Fig. 5.2-2 shows the structural arrangement of the LEM descent stage
- Fig. 5.2-3 shows the inboard profile of the LEM ascent stage
- Fig. 5.2-4 shows the general arrangement of the LEM descent stage
- Fig. 5.2-5 shows the general arrangement of the LEM vehicle

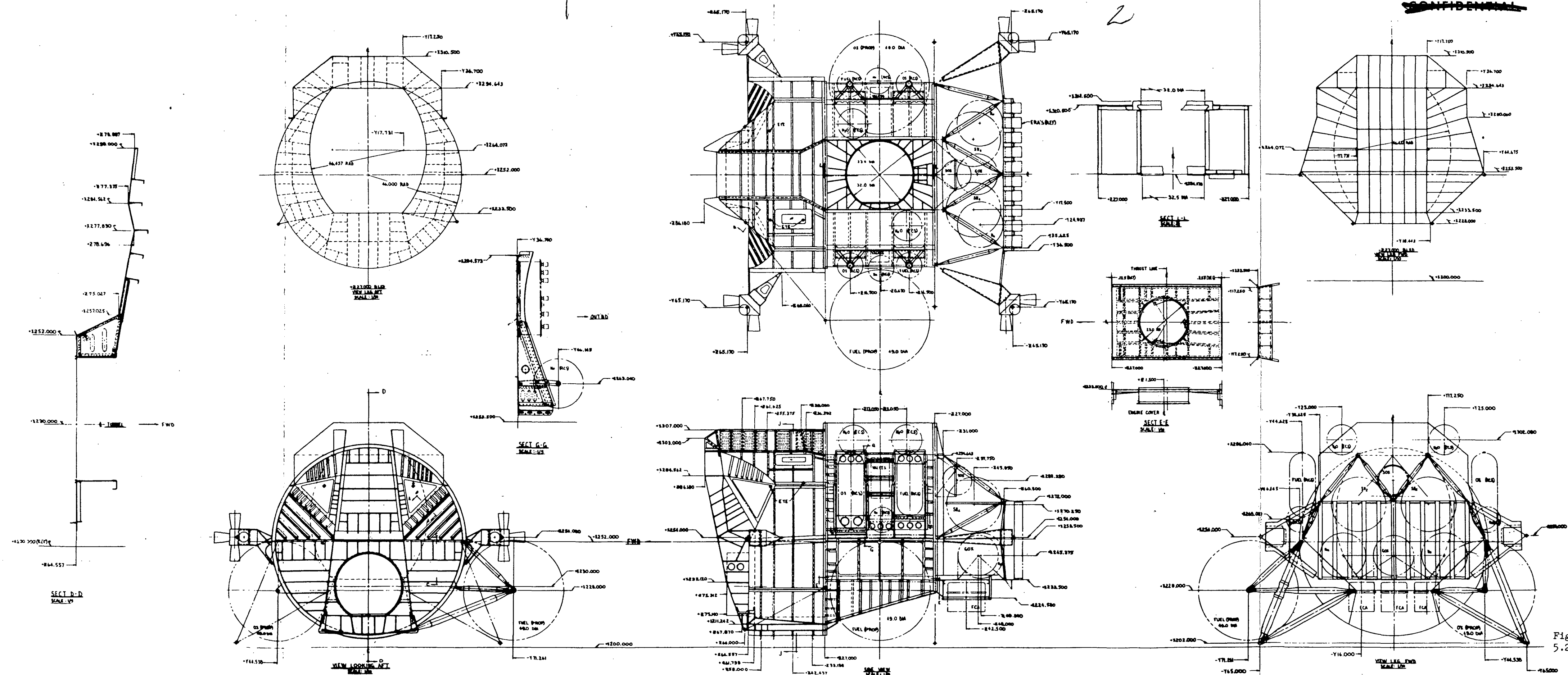


Fig.
5.2-1

Grumman
AIRCRAFT ENGINEERING CORPORATION
FELDBURNER DESIGN BLDG.
Strcut Arrgmt
Ascent Stage
Sheet 2 of 2
SCALE OF DIM. INCH. -
AES1001

NOTE:
AES1002 - IDENTICAL TO STRUCTURAL
ARRANGEMENT DESCENT STAGE
DRAWING #LDW 280-21600 9-19-68

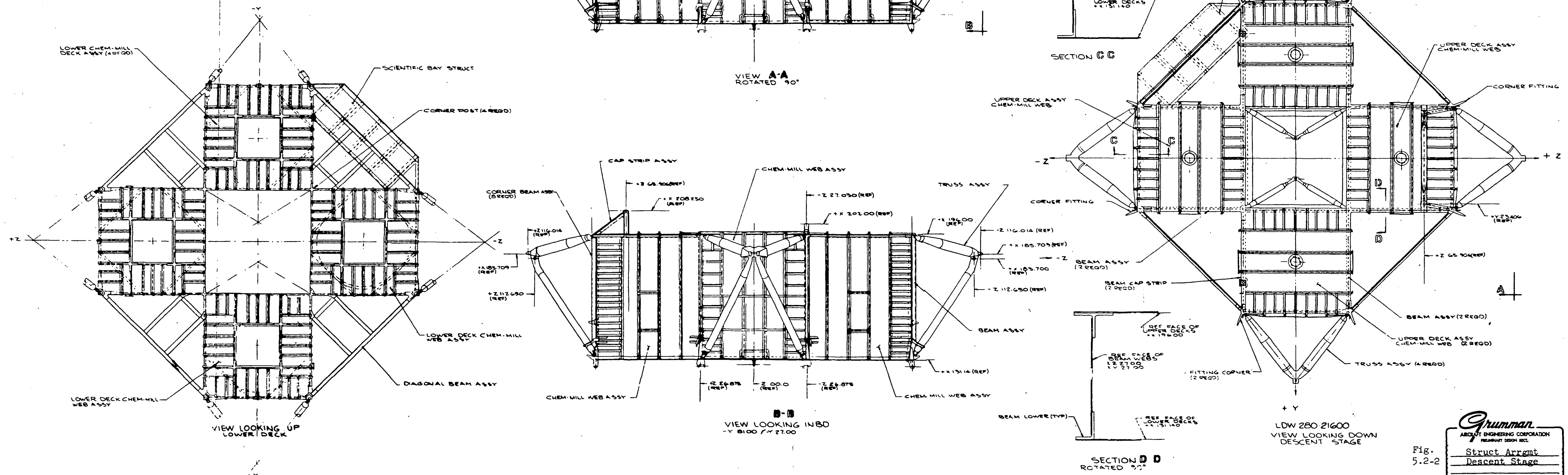


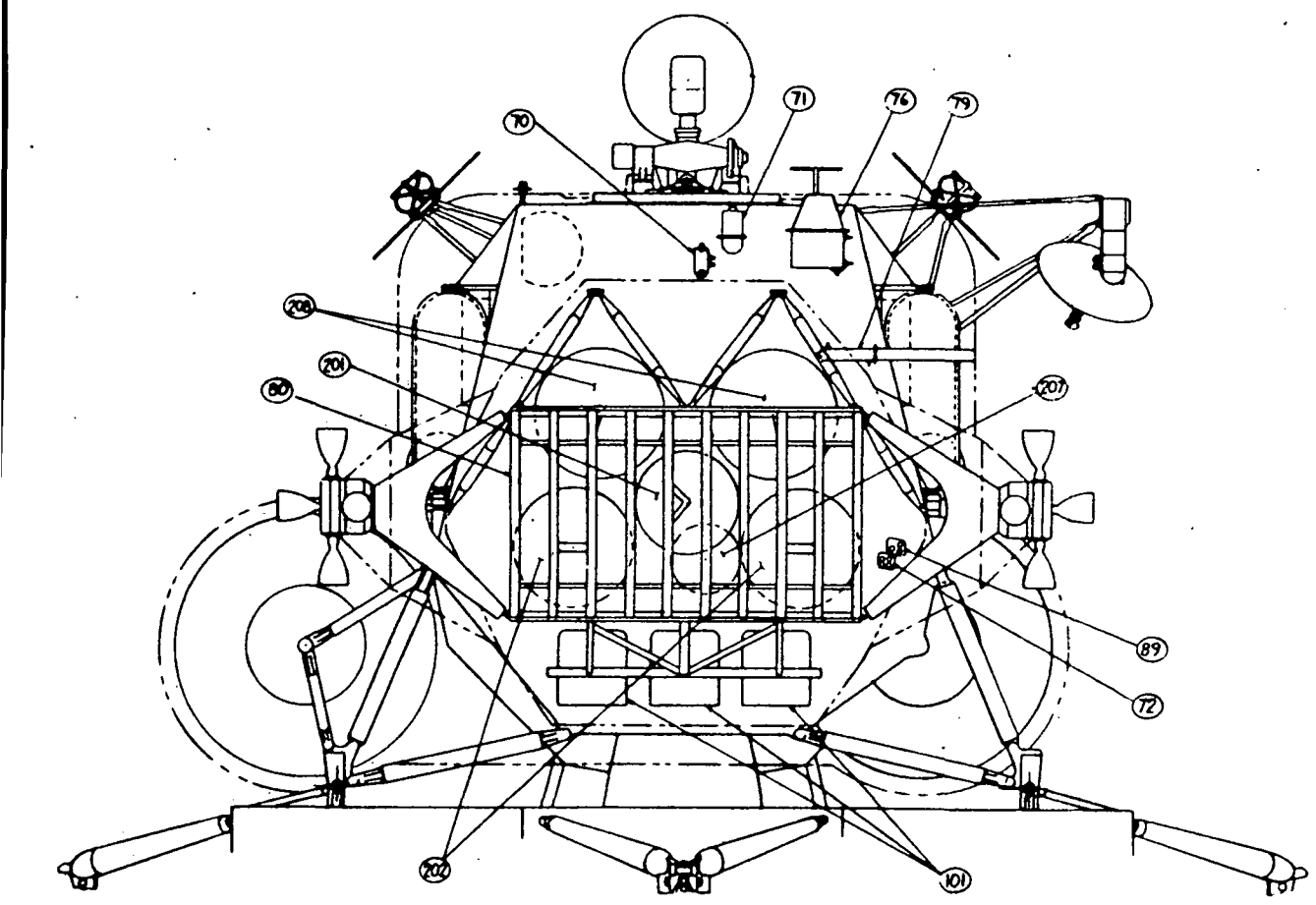
Fig. 5.2-2

Grumman
AIRCRAFT ENGINEERING CORPORATION
PRELIMINARY DESIGN DEPT.

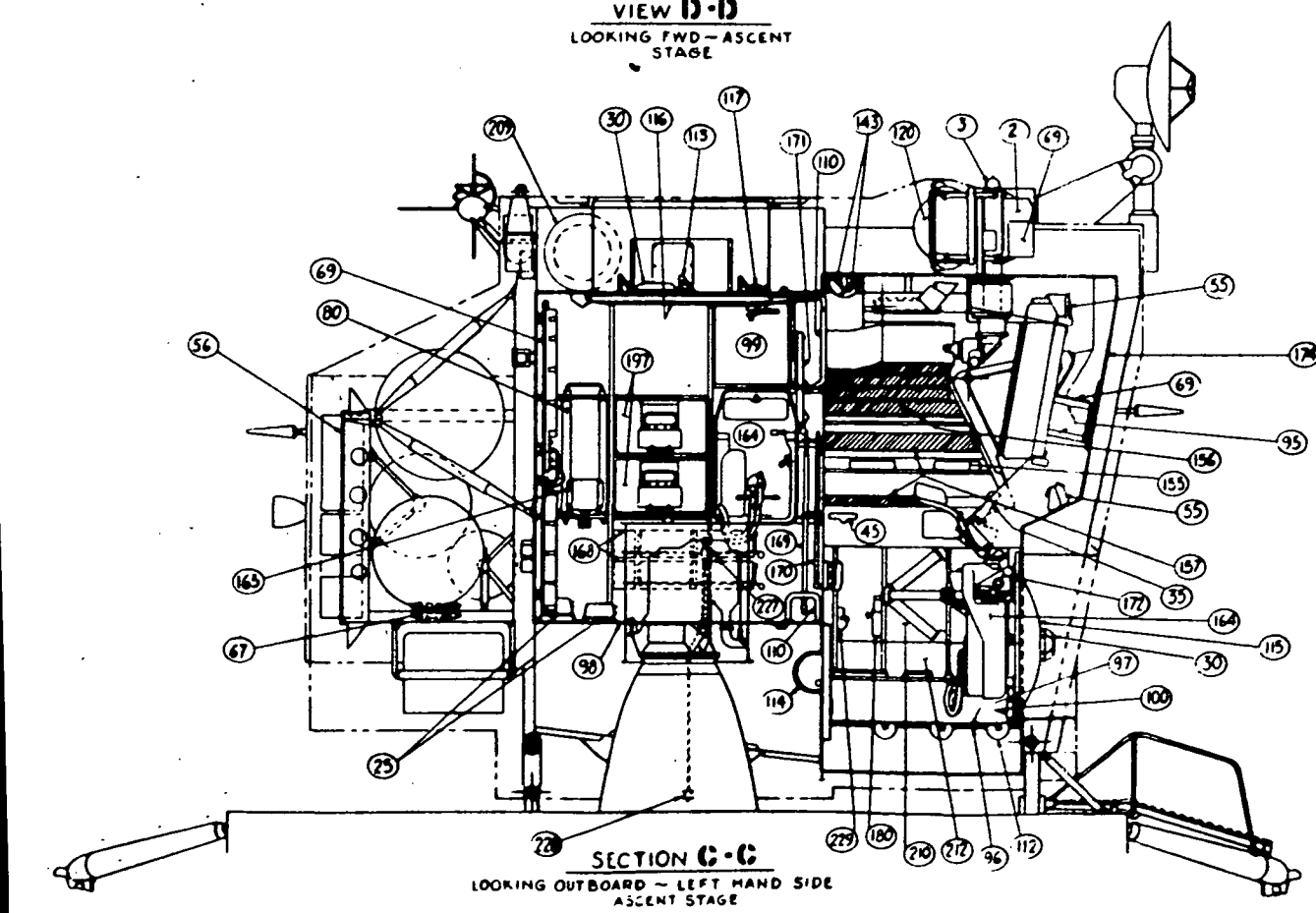
Struct Arrmt
Descent Stage

SCALE OF ORIG. DWG. =

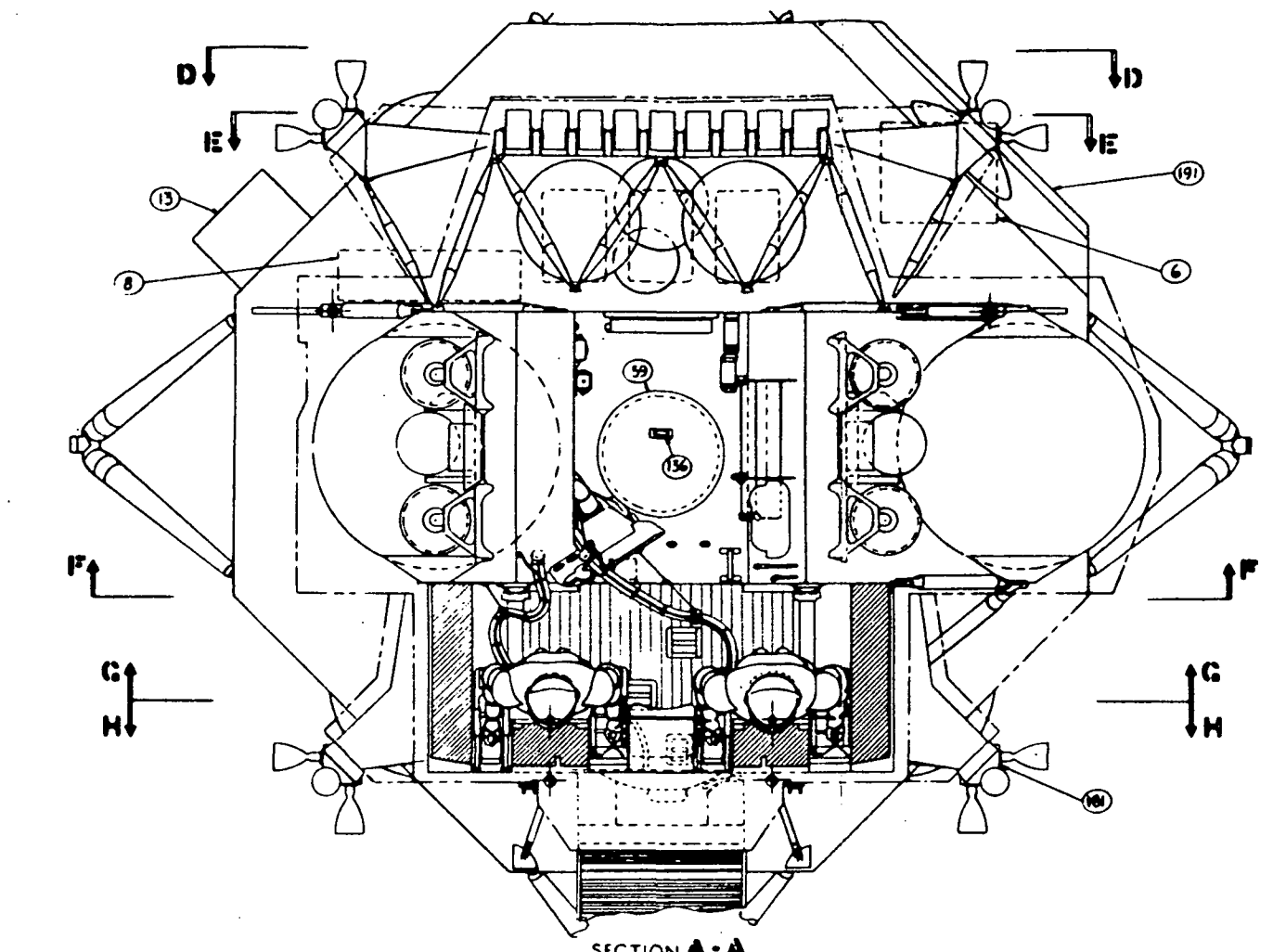
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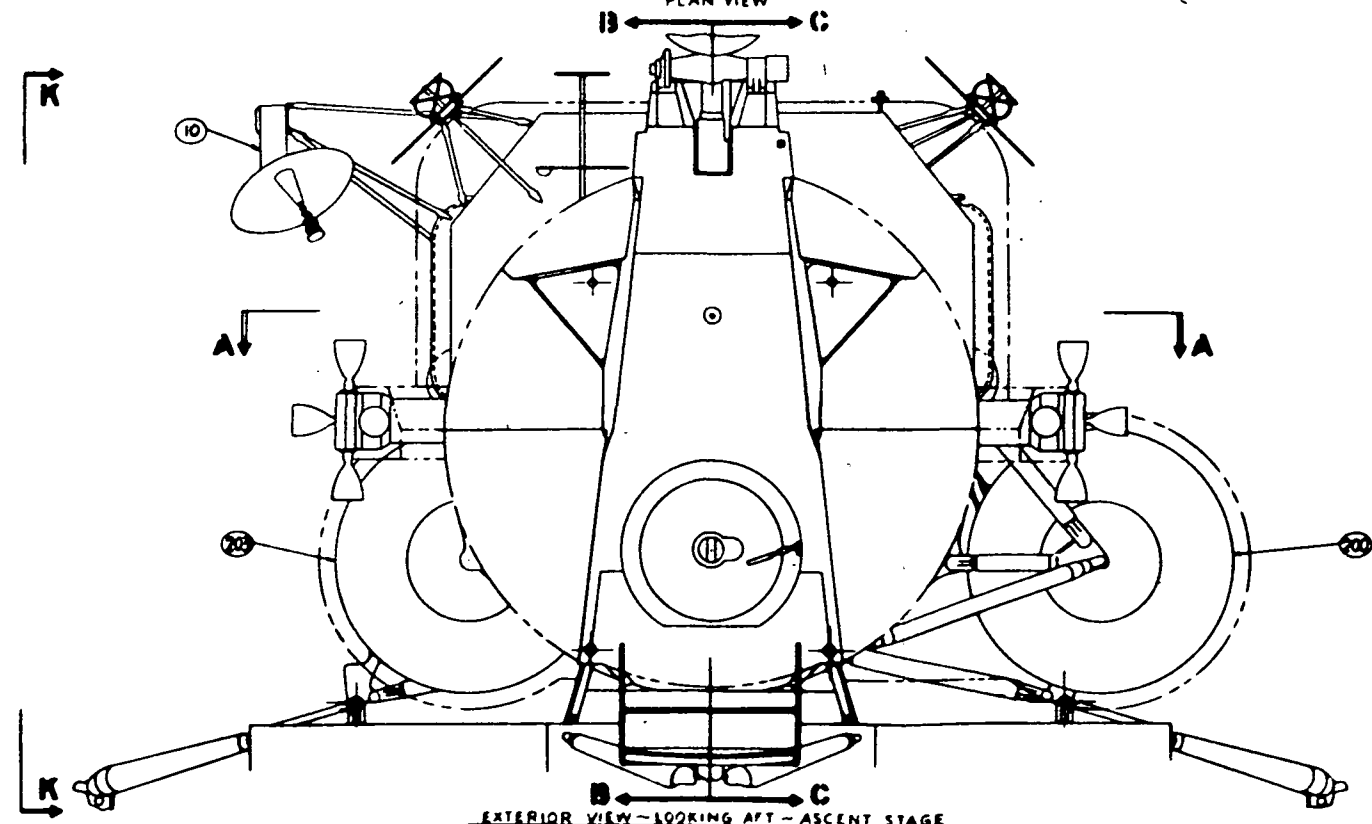
VIEW D-D
LOOKING FWD-ASCENT
STAGE



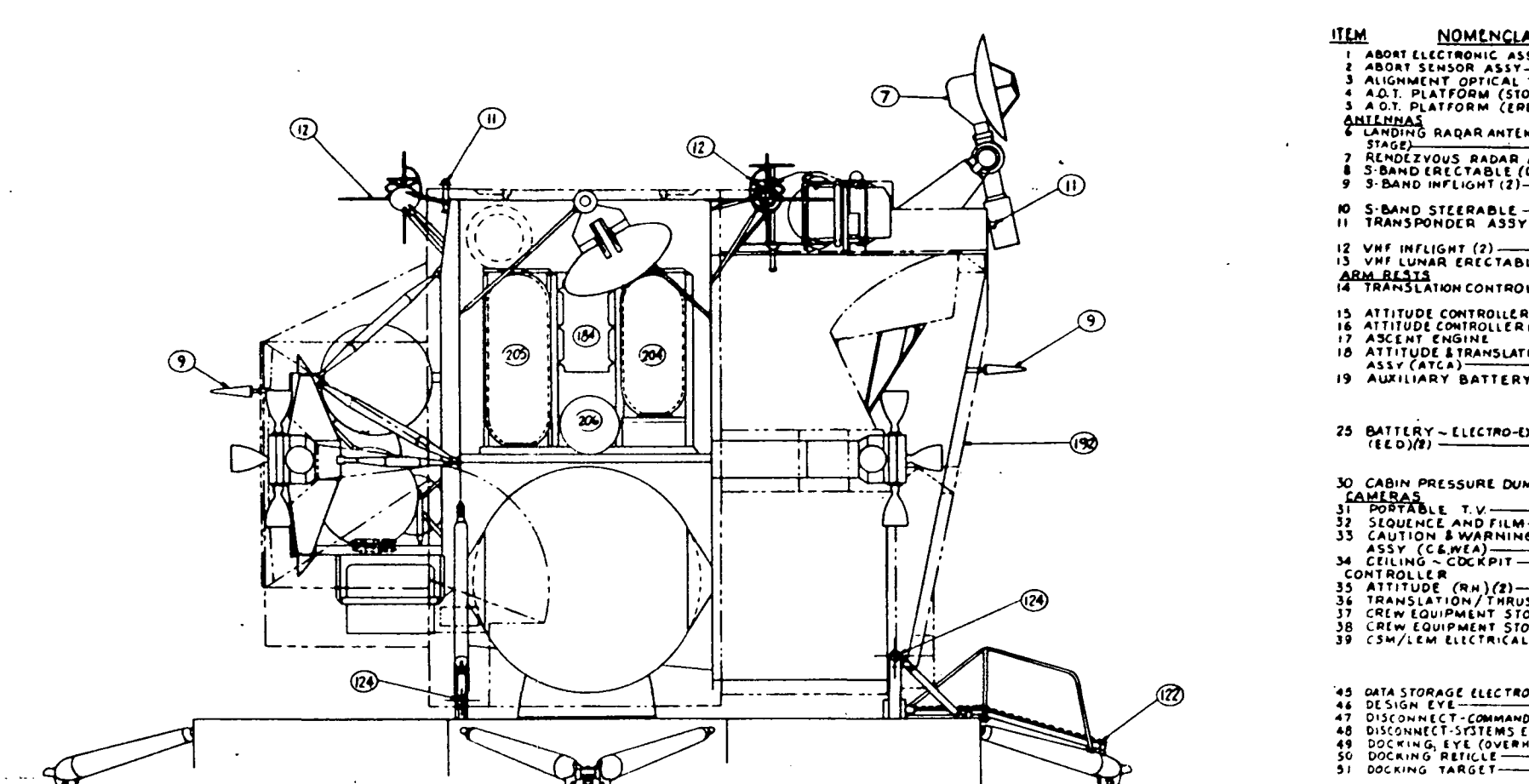
SECTION C-C
LOOKING OUTBOARD - LEFT HAND SIDE
ASCENT STAGE



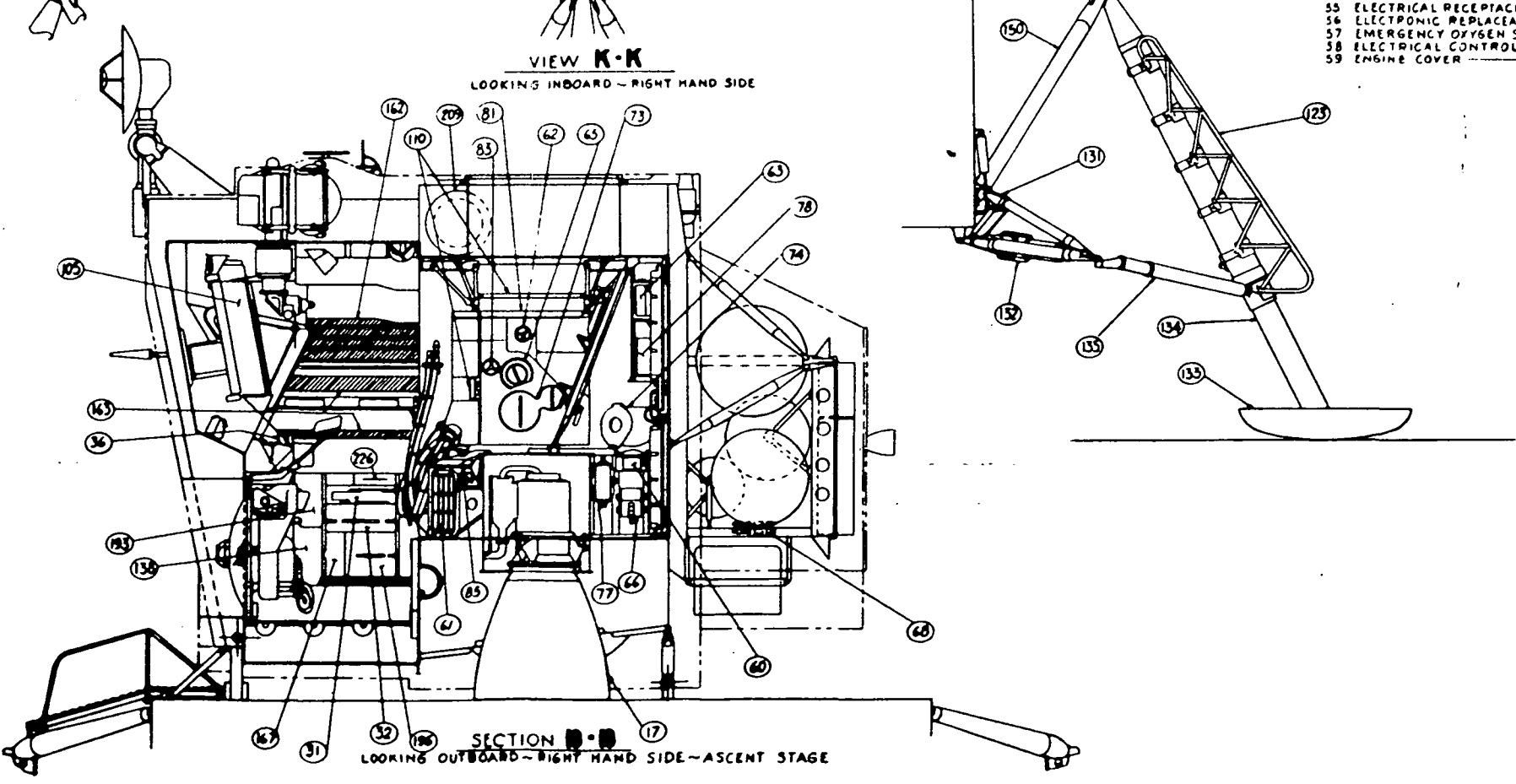
SECTION A-A
PLAN VIEW
LOOKING INBOARD - RIGHT HAND SIDE



EXTERIOR VIEW - LOOKING AFT - ASCENT STAGE



VIEW K-K
LOOKING INBOARD - RIGHT HAND SIDE



SECTION B-B
LOOKING OUTBOARD - RIGHT HAND SIDE - ASCENT STAGE

ITEM	NOMENCLATURE	ZONE	MPC	DESIG	ITEM	NOMENCLATURE	ZONE	MPC	DESIG
1	ABORT ELECTRONIC ASSY (AEA)	13C	2.4		164	PORTABLE LIFE SUPPORT SYSTEM (PLSS) (2)	11A	6.1	
2	ABORT SENSOR ASSY	10B	2.1		165	PLSS - BATTERY CHARGER	11A	6.1	
3	ALIGNMENT OPTICAL TELESCOPE (AOT) (1)	10B	3.1		166	PLSS - DRAIN/RECHARGE STATION	11A	6.1	
4	A.O.T. PLATFORM (STOWED)	15A			167	PLSS - SUPPORT - ERECTED	16A	4.1	
5	A.O.T. PLATFORM (ERECTED)	15A			168	PLSS - DRAIN/RECHARGE STATION - SUPPORT	16A	4.1	
6	LANDING RADAR ANTENNA ASSY (DESCENT STAGE)	7C	3.3		169	PLSS - LOW CHARGE (6)	11A	6.1	
7	RENDZVOUS RADAR ASSY	4C	3.2		169	PLSS - RECHARGE HOSE (40')	11A		
8	S-BAND ERECTABLE (DESCENT STAGE)	9C	12.3		170	PLSS - RECHARGE HOSE (10')	11A		
9	S-BAND INFIGHT (2)	9C	11.2		171	PLSS - SUPPORT IN EQUIPMENT TUNNEL	11A		
10	S-BAND STEERABLE	9C	12.3		172	PLSS - SUPPORT ON FRONT MATCH	11A		
11	TRANSPONDER ASSY	10B	2.4		173	POWER SERV ASSY & ELECTRONIC	11A		
12	VHF INFIGHT (2)	9C	11.2		174	SCURLING DISPLAY UNIT (PSA & ECU)	15C		
13	VHF LUNAR ERECTABLE (DESCENT STAGE)	9C	11.3		174	PRESSURE SHELL OUTLINE	10A		
14	ARM RESTS	15C			175	PROPELLANT QUANTITY MEASUREMENT SYSTEM	11A	3.7	
15	TRANSLATION CONTROLLER (L.H.) (2)	10C	4.3		176	PULSE CODE MODULATION AND TIMING EQUIPMENT (PCMT)	15C	1.1	
16	ATTITUDE CONTROLLER (RH) SYSTEMS ENGR	10C	4.3						
17	ATTITUDE CONTROLLER (RH) COMMANDERS	10C	4.3						
18	ASCENT ENGINE	7A	9.1						
19	ATTITUDE & TRANSLATION CONTROL ASSY (ATCA)	13C	2.2						
20	AUXILIARY BATTERY	13C							
21	BATTERY - ELECTRO-EXPLOSIVE DEVICE (EED) (2)	11A	8.1						
22	CABIN PRESSURE DUMP VALVE (2)	10A							
23	CABIN PRESSURE DUMP VALVE (2)	10A							
24	SEQUENCE AND FILM	6A	7.3						
25	CAUTION & WARNING ELECTRONIC ASSY (C&WA)	13C	7.3						
26	CEILING - COCKPIT	10C	1.2						
27	CONTROLLER	10A							
28	ATTITUDE (RH) (2)	10A	12.1						
29	TRANSLATION/THRUST (L.H.) (2)	10A	12.1						
30	CREW EQUIPMENT STORAGE (L.H. SIDE)	10B	1.9						
31	CREW EQUIPMENT STORAGE (R.H. SIDE)	10B	1.9						
32	CSM/LEM ELECTRICAL UMBILICAL CONNECTORS	15C	8.4						
33	DATA STORAGE ELECTRONICS ASSY (DSEA)	11A	7.1						
34	DESIGN EYE	13A							
35	DISCONNECT - COMMANDER'S POWER & COMMUN.	15C							
36	DISCONNECT - SYSTEMS ENGR'S POWER & COMMUN.	15C							
37	Docking, eye (overhead)	10A							
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190	Docking,								

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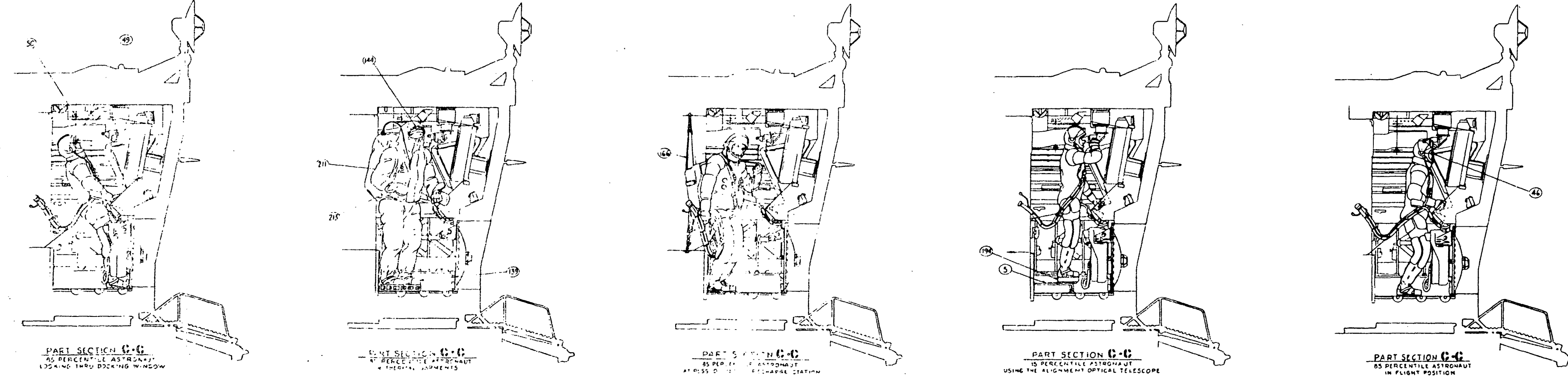
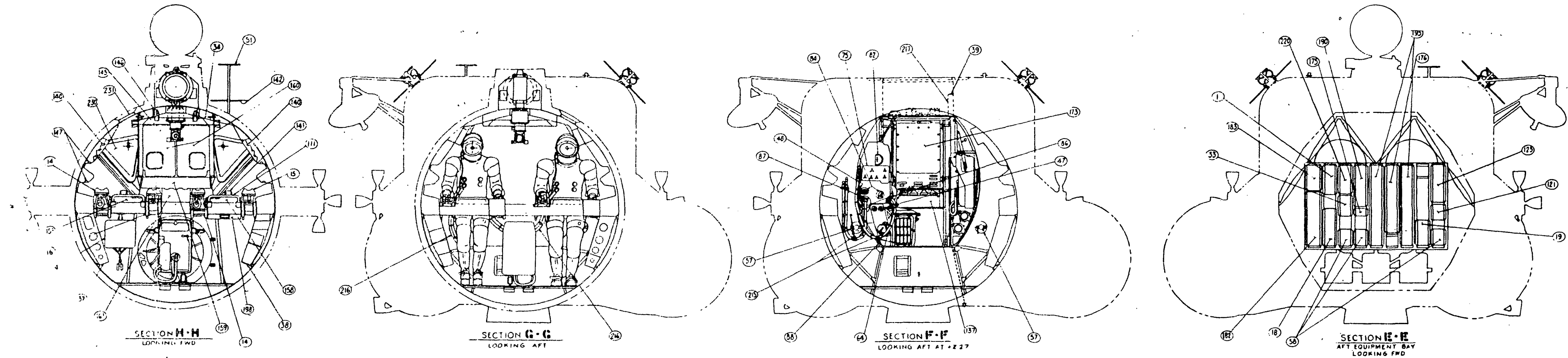


Fig. 5.2-3

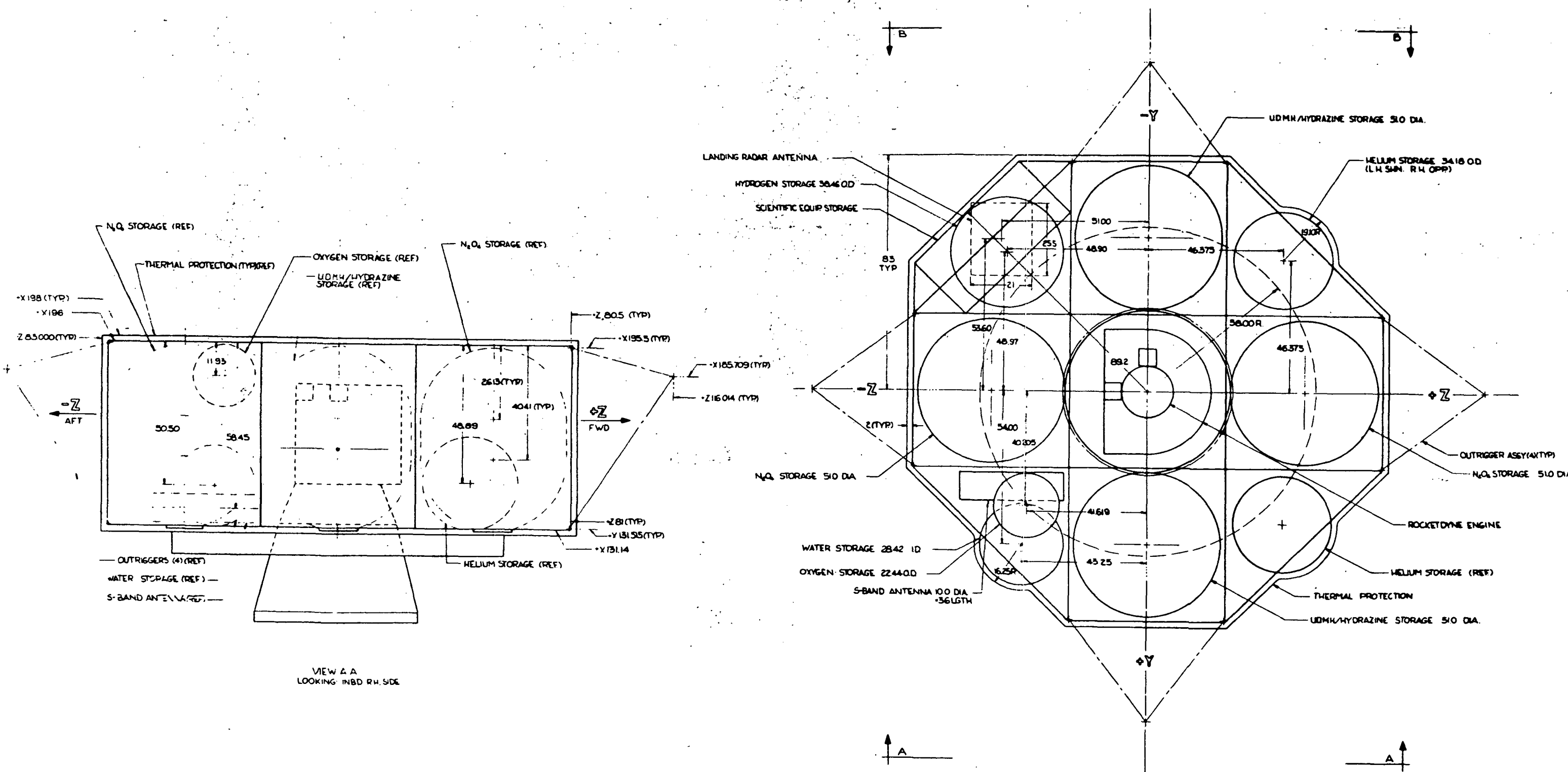
Grumman
AIRCRAFT ENGINEERING CORPORATION
PREPARED BY DESIGN DIV.

Inbd Profile
Ascent Stage
Sheet 2 of 2

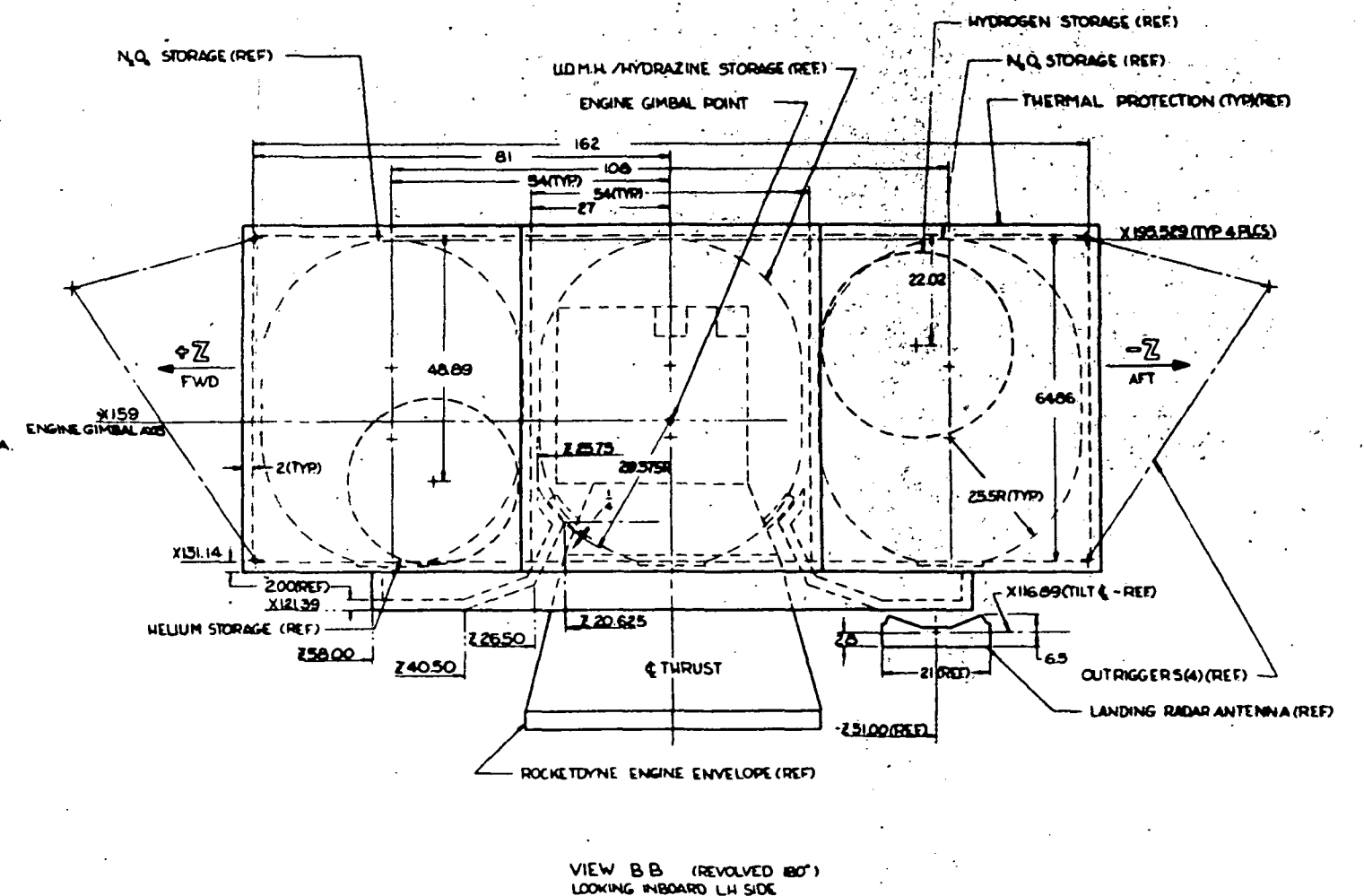
SCALE OF ORIGIN: 1/2" = 1'-0"

AES1003

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NOTE:
 AES1004 - IDENTICAL TO GENERAL
 ARRANGEMENT DESCENT STAGE DRAWING
 #LDW 280-17000. 3-6-63



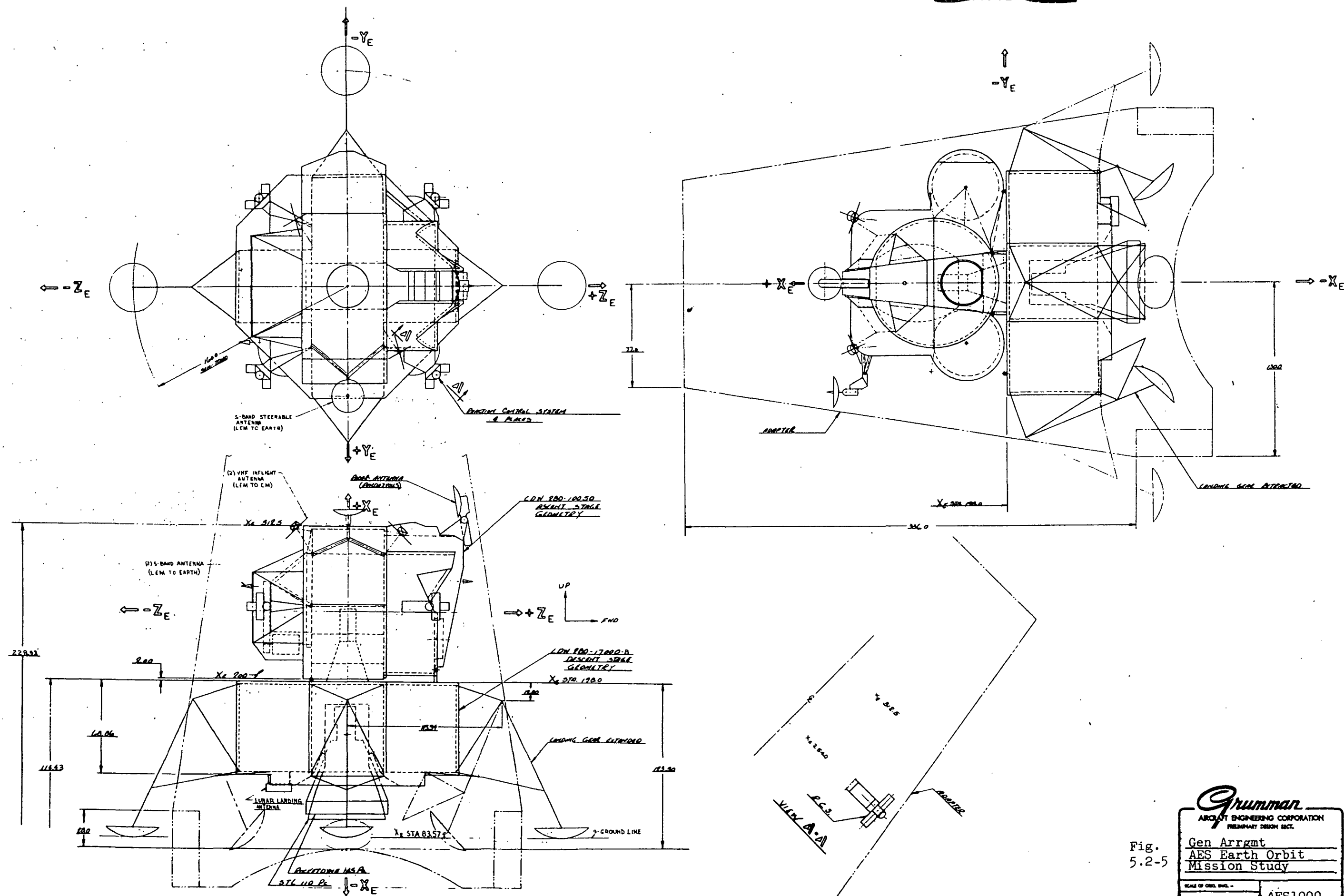


Fig.
5.2-5

Grumman
AIRCRAFT ENGINEERING CORPORATION
PRELIMINARY DESIGN SECT.

Gen Arrgmt
AES Earth Orbit
Mission Study

SCALE OF ORBIT SWG. -

AES1000

5.3 STANDARDIZED CONFIGURATION MODULES

Two standardized configuration arrangements were investigated in this study.

- (1) Low Profile Descent Stage
- (2) Power Package Installations

Figure 5.3-1 shows the low profile descent stage (30" depth) that was selected as a standard supporting structure for experiments that require additional volume beneath the existing LEM descent stage. The structural arrangement is identical to the LEM descent stage with the exception of the reduced depth, and can be fabricated in the existing LEM descent stage assembly fixture.

The 30" depth was selected as the best compromise considering the following:

- (1) Total lab weights varying between 8,000 - 20,000 lbs; with the majority of flights under 12,000 lbs.
- (2) Low profile descent stage structural weight
- (3) Volume requirements for experiments

Structural analyses associated with the selection of the 30" D/S are discussed in Section 4.3.

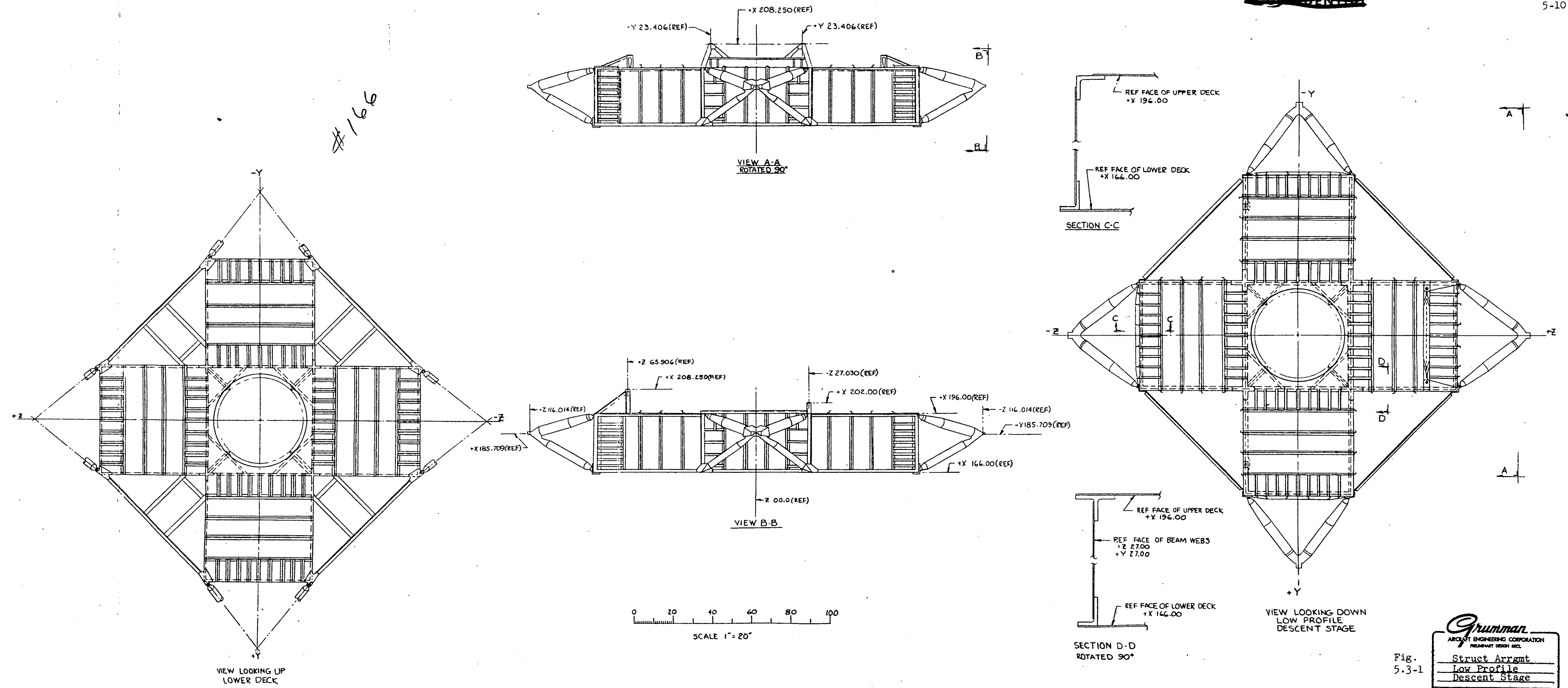
The low profile descent stage is used on NASA Flights 211, 509, 219, 521, 229, and DOD Flights 4, 5, 6, 7A, 7B and 10.

Lab flights that require an airlock or passageway in the descent stage (219, 229, 230) require additional support structure in the center bay of the descent stage. Fig. 5.3-1 also shows the type of support structure required, for either 30" or full depth descent stages, to accommodate the passageway.

Standardized power package configurations that were evolved, during this study, are shown in Figures 5.3-2 thru 5.3-4.

Configuration I (Figure 5.3-2) uses Block II H_2 and O_2 tanks. Configuration II (Figure 5.3-3) uses Apollo X H_2 and O_2 tanks. Configuration III uses combinations of Block II and Apollo X tanks. Fuel cell locations are common to all configurations.

These arrangements represent the basic power pack configurations to/from which tankage and fuel cells are added or subtracted depending upon mission requirements. Figures 5.3-2 thru 5.3-4 identify their respective flight allocations.



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Grumman
AIRCRAFT ENGINEERING CORPORATION
PRELIMINARY DESIGN SKETCH

Fig. 5.3-1

Struct Arrgmt
Low Profile
Descent Stage

AES1005

Flight	Oxygen Tanks	H ₂ Tanks	F.C.A.
211C	1 III	1 III	2
D	1 III	1 III	3
	1 II	1 II	
218C	1 III	1 III	2
221E	1 III	1 III	2
516C	1 III	1 III	2
230C	1 III	1 III	2
		1 II	
DOD 1C	1 III	1 III	2
1D	1 III	1 III	3
		1 II	
IIC	1 III	1 III, 1 II	3
IIIC	2 II	1 III	2
IIID	1 III	1 III	3
523C	1 III	1 III	2
211A-C	1 III	1 III	2
-D	1 II	1 II	
	1 III	1 III	3
DOD-7A	1 III	1 II	5
		1 III	

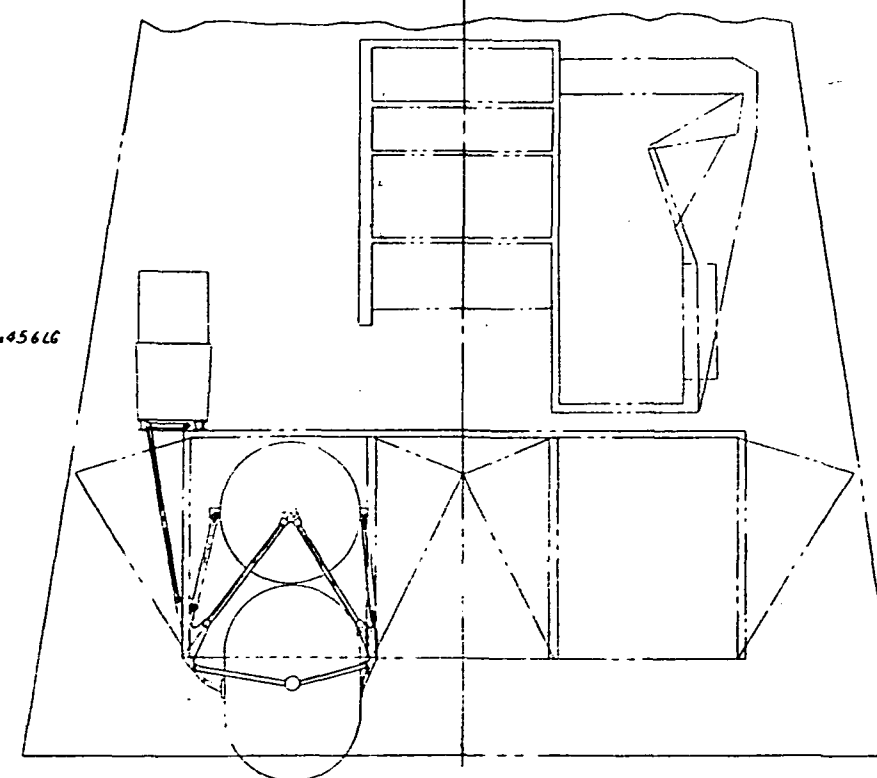
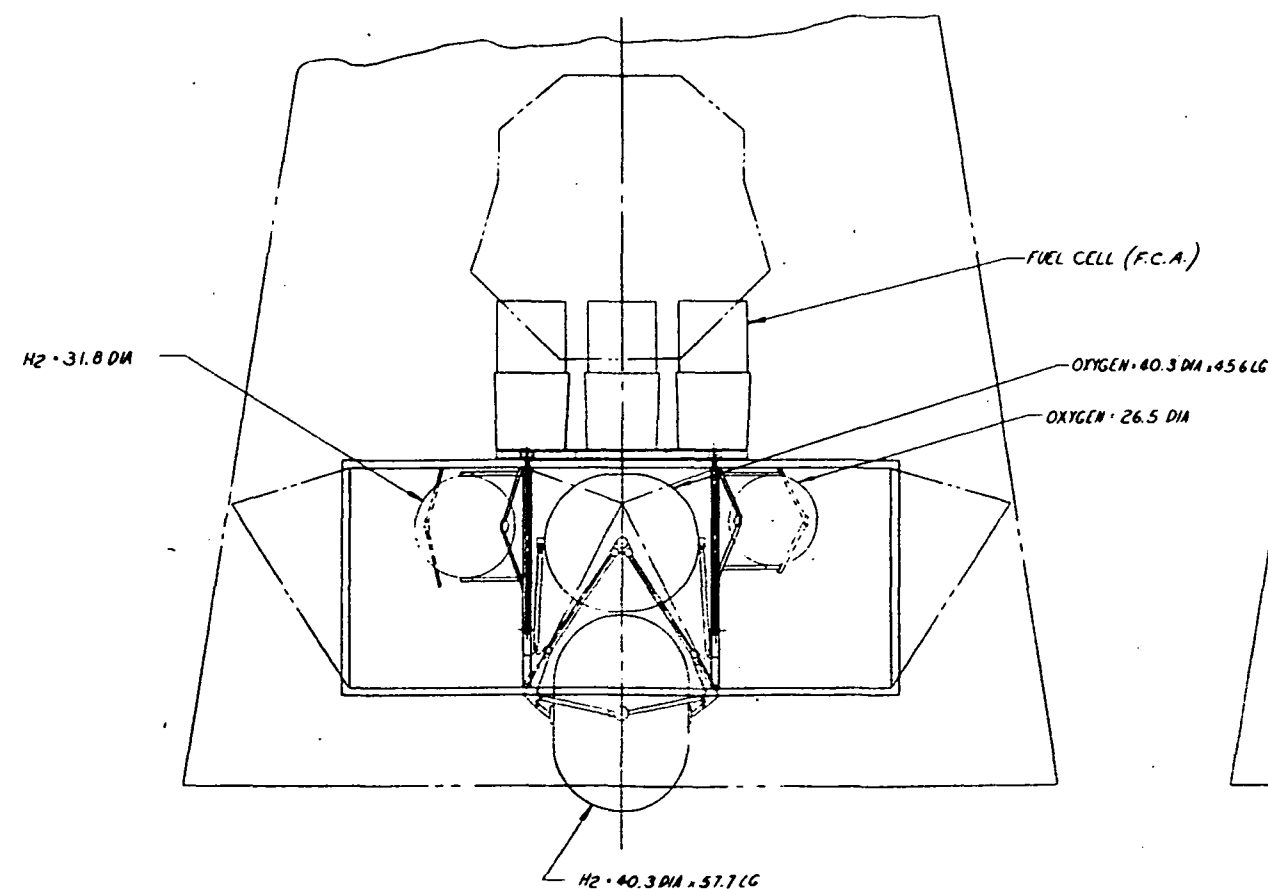
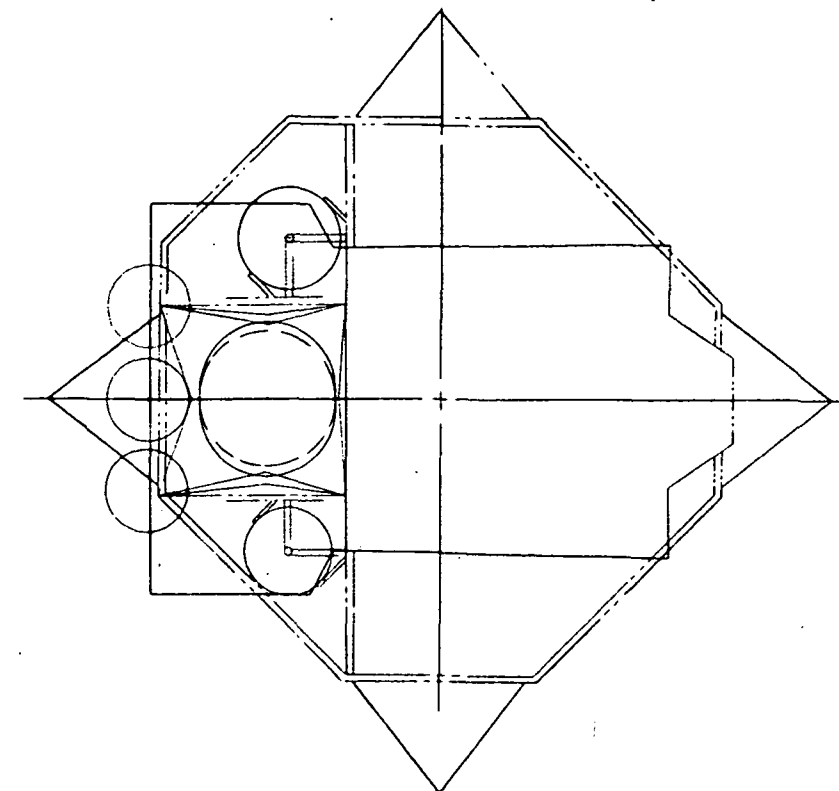


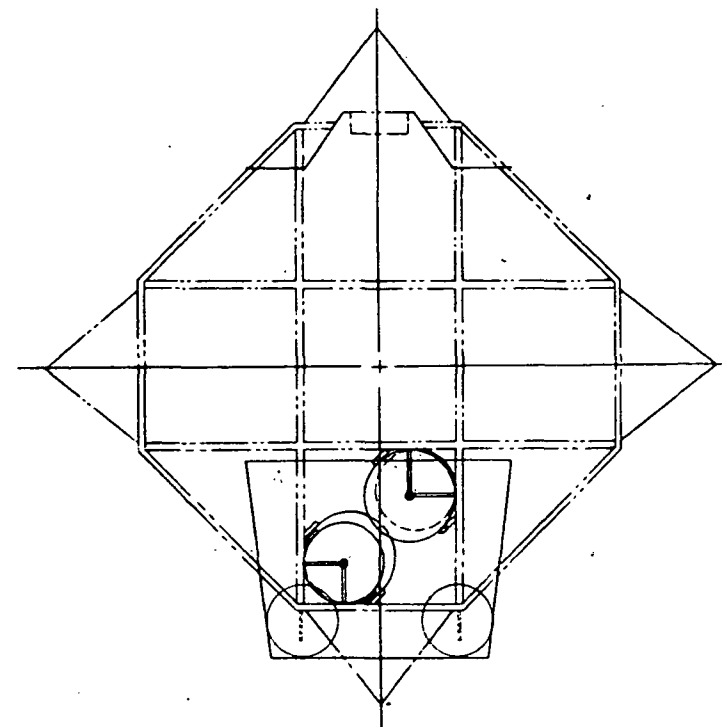
Fig.
5.3-2

Grumman
AIRCRAFT ENGINEERING CORPORATION
PRELIMINARY DESIGN SECT.

Struct Arrgmt Power
Pack Config I

SCALE OF CHG. DWG. - 1/20

AES 1006



Flight No.	Oxygen Tanks	H ₂ Tanks	F.C.A.
507	1 II	1 II	2
509	1 II	1 II	2
215	2 II	2 II	2
513	2 II	2 II	2
DOD 4	1 II	1 II	2
DOD 5	1 II	1 II	2
DOD 6C	2 II	2 II	2

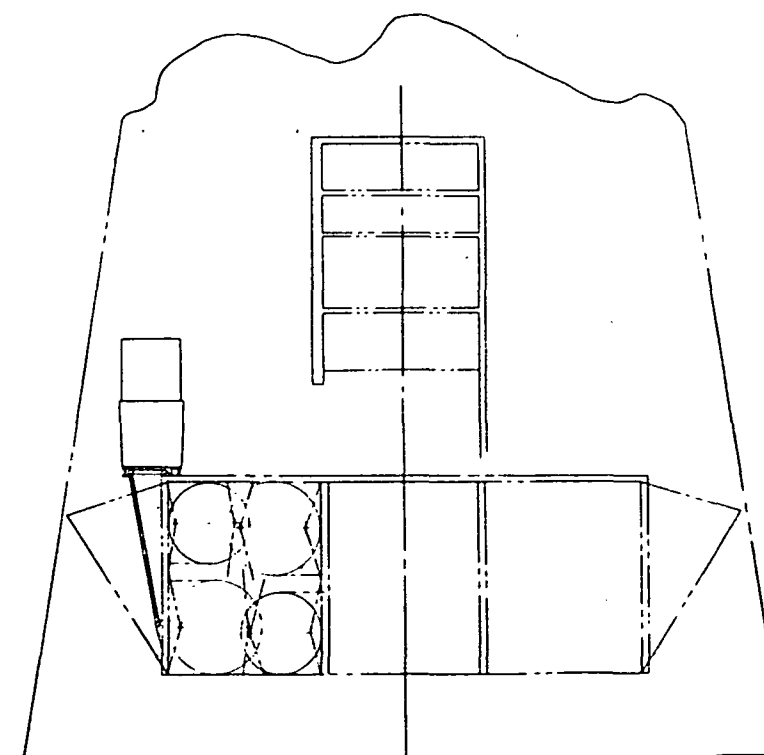
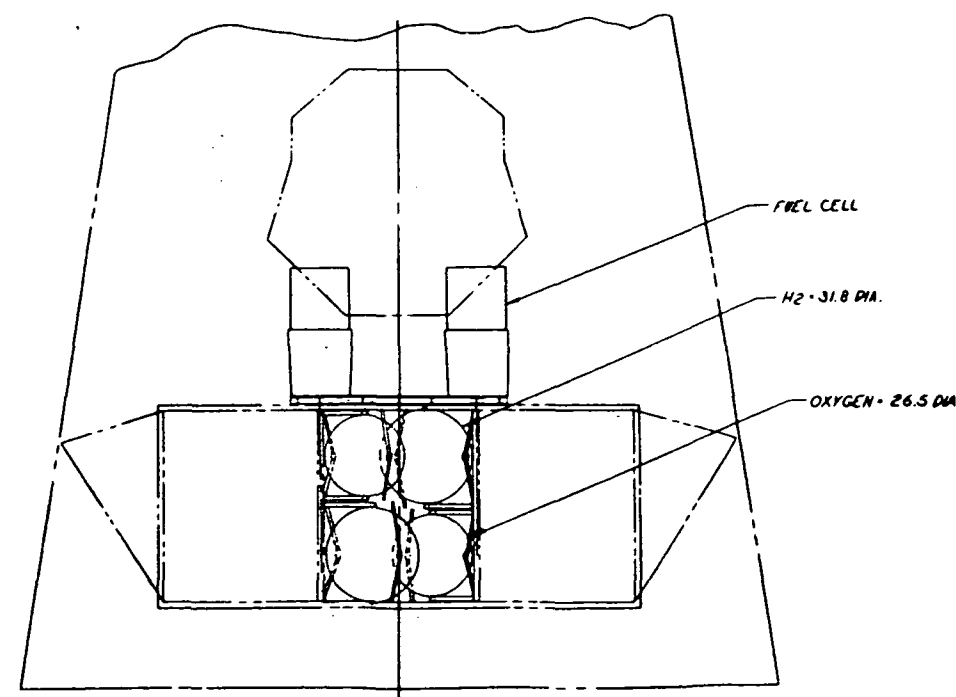


Fig. 5.3-3

Grumman
AIRCRAFT ENGINEERING CORPORATION
PRELIMINARY DESIGN SECT.
Struct Arrgmt
Power Pack Config II
SCALE OF ORG. Dwg. 1/20
AES 1007

Configuration III Power Pack
Used On Flight Nos:

218D
221F
516D
518D
521D
523D
230D
518C
521C
DOD-2D
DOD-8

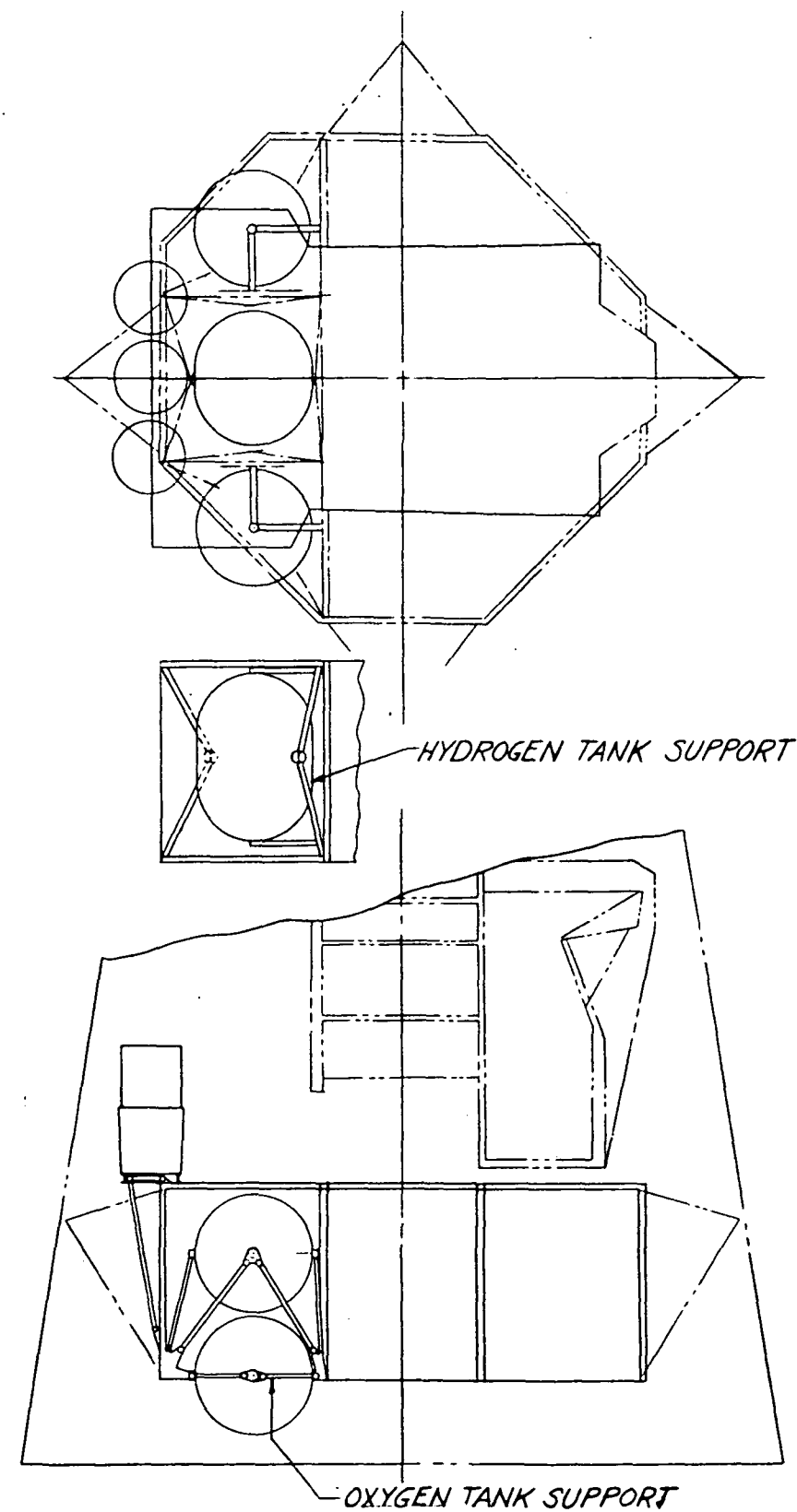
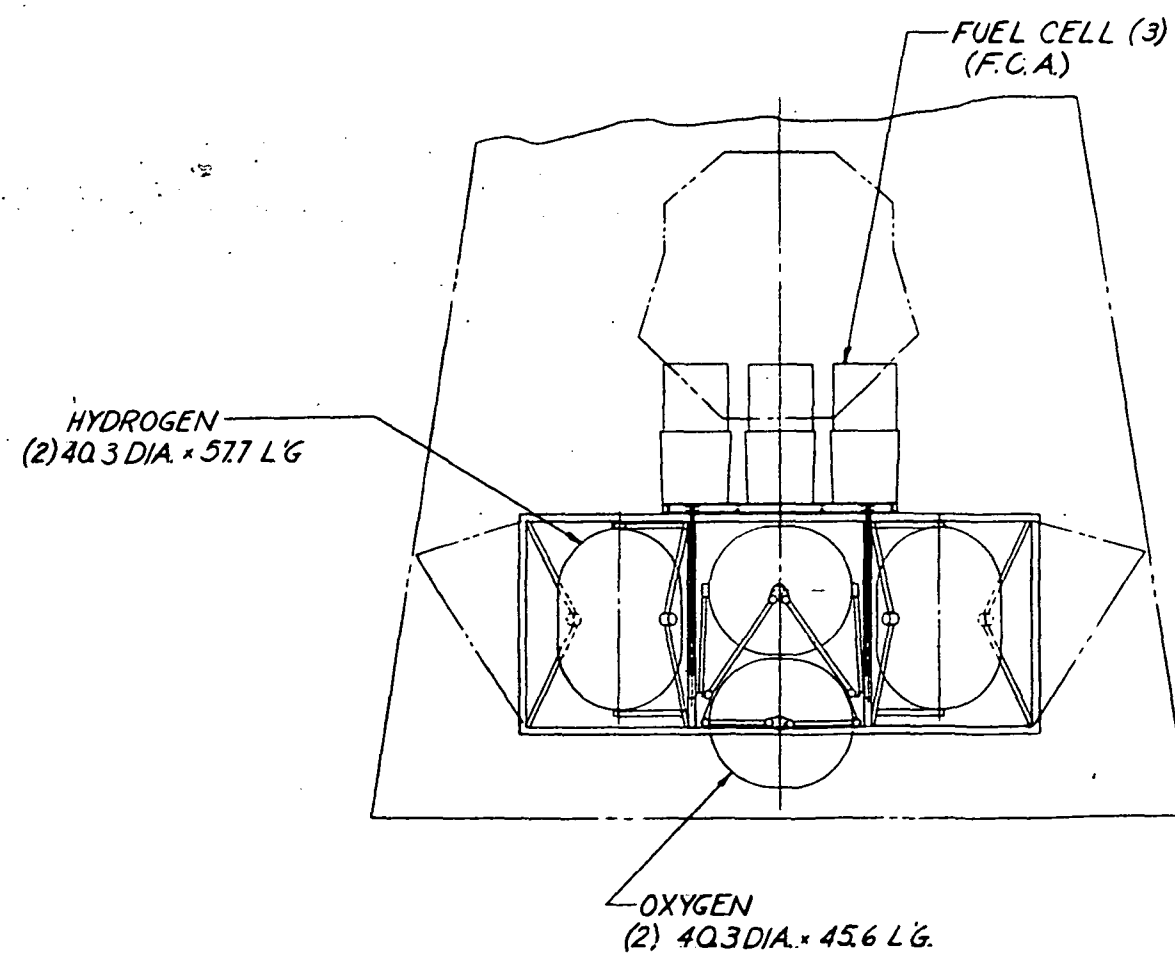
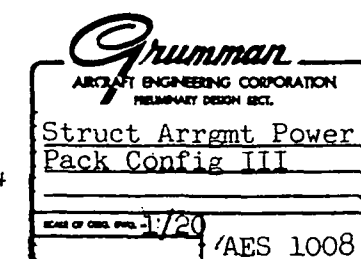


Fig.
5.3-4



~~CONFIDENTIAL~~

5.4 EXPERIMENT EQUIPMENT AND SPACECRAFT SUBSYSTEM COMMONALITY

A major design objective throughout the study was standardization of experiment and spacecraft subsystem installations.

Table 5.4-1 illustrates the amount of experiment equipment installation commonality achieved for the NASA flights. In general, a number of experiments are carried on more than one flight; in these cases, the equipment installations in Lab spacecraft are common.

Table 5.4-2 illustrates spacecraft subsystem commonality for NASA and DOD flights. Power supply, thermal control, RCS, and flight control systems are shown. Except for one DOD flight and two NASA flights, the standardized power pack configurations (with minor variations) (ref. Section 5.3) can be used.

Thermal control radiator sizes vary from zero to 190 ft², for equipment, and from zero to 88 ft², for FCA radiators. These variations verify the need for a modular radiator panel approach, as assumed for this study (Ref. sect. 3.2.1.4).

RCS systems and flight controls are carried on 14 of 27 NASA and DOD flights studied. Attitude reference sensors (horizon scanner, earth, and star trackers) are carried on 21 flights. For these flights, CSM control system interfaces vary from attitude sensor inputs (in 16 cases) to both sensor and attitude controller inputs (in 5 cases).

TABLE 5.4-1 EXPERIMENT EQUIPMENT COMMONALITY

FLIGHT	LICH CANNISTER SPARES-EXT. STUR.	EVA PLATFORMS & RAILS	FOOD COMP. FWD. FLOOR	BIO MED CANNISTER	ANTENNAS										CAMERAS														DEPLOYABLE BALLOONS-LINES-NETS-RODS-ETC.														MAGNETOMETER					
					MICROWAVE 0901	VHF PULSE 0902	MIC. RADIO 0902	HI-LOW FREQ. 1201	HI-FREQ. TRANS. 1202	VHF TRANS 1202	MILLIMETER 1202	ERECTABLE 1306	12-24-48 DISH 0806	RADAR IMAGERY 0903	16 M.M. 1500 1405	MULTI-SPEC LOS	CARTOGRAPHIC LOS	PAN LOS	MULTI-SPEC 0901	24" F. L. 0902	6" F.L. 0902	70 M.M. 0902-1603	SPECTRO 0503	35-70-M.M. 0503	DOCUMENT 0802	ORTHICON T. U. 0503	PAN 0903	SAFETY STATIC 1500	MOTOR TRANS. 1504	CARGO NET 1500	PERSONNEL TRANS 1503	50' BALLOON 1302;040	TARGET SATELLITE 1601	200' ROD 1303	CLOUD GEN 0503	HARD SUIT 1500	MANEUVER SATELLITE 1001	CERAMIC SPHERE 1101	MICRO BALLOON 0504	1000 TUBE 1303	MAG. LOS	MAG 0801	MAG 0502	MAG 0606				
209	•	•	•	•							•															•		•	•																			
209 A	•	•	•	•							•															•		•	•																			
211C D	•		•	•								•	•	•																								•	•	•	•							
211A-C -D	•		•	•								•	•	•																								•	•	•	•							
507	•		•	•	•	•	•	•	•	•	•				•		•	•	•																													
509	•		•	•																																												
215	•		•	•	•	•									•		•	•	•																													
218C D	•		•	•							•	•															•	•	•	•																•	•	
219C D		•																									•	•																				
513	•		•	•							•								•	•	•	•	•																									
221C D	•		•	•																																												
516C D	•																																															
518C D	•		•	•	•	•					•				•	•	•	•	•						•	•																						
521C D	•		•	•															•	•	•	•																					•	•	•	•		
523C D	•																																															
229C D	•																																															
230C D	•																																															
16																																																

• COMMON EQUIPMENT INSTALLATIONS

○ UNIQUE EQUIPMENT INSTALLATIONS

STORED IN C/M IN FLIGHT-IN LEM AT LAUNCH

TABLE 5.4-1 (Cont'd) EXPERIMENT EQUIPMENT COMMONALITY

FLIGHT	RADAR			SECTROMETERS									TELESCOPE-TRACKERS-VIEWFINDERS									MECHANICAL EQUIPMENT	
	MAPPING LOS	ALTIMETER LOS	ALTIMETER 1407-0903	UV LOS	IR SURVEY LOS	GRATING 0901	PHOTOMETER 0901	IR SPEC & SCAN. 0802	MICROWAVE 0802C	PRO-ALPHA 0501	HI ENERGY 0501	LO ENERGY 0501	VIEW FINDER 0902, 0802 LOS	16" TELESCOPE 0901	TELESCOPE ASSY 0801	STAR TRACKER 0802, 1407	SMALL TELESCOPE 1407	MANNED CORONA-GRAPH 0703	INT'MED REFLECT TELESCOPE 0702	60" TELESCOPE 1406	RADIO-ASTRONOMY 0705A	SOLAR CELL ARRAY 1405	ARTIFICIAL G MECHANISM 0301
209										●	●	●											
209 A										●	●	●											
211C	●	●		●	●					●	●	●	●										
D	●	●		●	●					●	●	●	●										
211A-C	●	●		●	●					●	●	●	●										
D	●	●		●	●					●	●	●	●										
507						●	●			●	●	●	●	●								●	
509										●	●	●			●								
215						●	●			●	●	●	●	●									●
218C			●							●	●	●					○						
D			●							●	●	●					○						
219C																							
D																							
513								○	○	●	●	●	●										
221C										●	●	●											●
D										●	●	●										●	
516C										●	●	●			●			●	●				
D										●	●	●			●			●	●				
518C			●							●	●	●	●									●	
D			●							●	●	●	●								●		
521C										●	●	●		●	●					○			
D										●	●	●		●	●					○			
523C										●	●	●						●	●		○		
D										●	●	●						●	●		○		
229C																							
D																							
230C																							
D																							
16																							

● COMMON EQUIPMENT INSTALLATIONS

○ UNIQUE EQUIPMENT INSTALLATIONS

TABLE 5.4-2 SUB-SYSTEM COMMONALITY DATA

Flight	POWER SUPPLY				FCA (No.)	Power Pack Config.**	THERMAL CONTROL		RCS SYSTEM		LAB Pointing Display & Control	LAB Flt. Control Sta.		FLIGHT CONTROLS		Nominal Vehicle Orientation
	Tankage						Radiate. Area, Sq. Equip.	FCA	RCS Jets	RCS Tankage Sets*		Attitude Contr. Sensors On LAB	LAB/CSM Interface With CSM Contr. Syst.			
	Oxygen		Hydrogen													
	Blk II	Ap.X	Blk II	Ap.X												
209	1						0	0	•	1		•	•	-	-	None
209-A	1						0	0	•	1		•	•	-	-	None
211 C		1		1	2	I-	42	15.1						Horizon & Star Trkrs.	Sensor inputs	Earth
D	1	1	1	1	3	I	46	30.8								Earth
211A-C		1		1	2	I-	42	15.1						Horizon & Star Trkrs.	Sensor inputs	Earth
-D	1	1	1	1	3	I	46	30.8						Horizon & Star Trkrs.	Sensor inputs	Earth
507	1		1		2	II-	70	29.2						Horizon & Star Trkrs.	Sensor inputs	Earth
509	1		1		2	II-	20	10	•	2		•	•	Earth & Star Trackers	Sensor inputs	Earth
215	2		2		3	II	60	20.6	•	1		•	•	Horizon & Star Trkrs.	Sensor inputs	Earth
218 C		1		1	2	I-	22	7.53	•	1	•	•	•	Horizon & Star Trkrs.	Sensor inputs	Earth
D		2		2	3	III	24	26.2	•	2	•	•	•	From Exp. 1407		
219 C	1	1		1	2	Special	28	11.3								Earth
D	2	2		2	3		30	30.6	•	2						
513	2		2		2	II	48	9.8						Horizon & Star Trkrs.	Sensor inputs	Earth
221 C		1		1	2	I-	38	7.85	•	2		•	•	-	-	Plane of
D		2		2	3	III	38	23.2	•	2		•	•			Orbit
516 C		1		1	2	I-	38	15.6			•	•	•	Earth & Star Trackers	Sensor & Controller	Inertial
D		2		2	3	III	40	36.2	•	1	•	•	•	inputs		Inertial
518 C		1		1	2	III-	60	14.5			•	•	•	Horizon & Star Trkrs.	Sensor & Controller	Earth
D		2		2	3	III+	64	29	•	1	•	•	•	inputs		
521 C		1		1	2	III-	18	8.56			•	•	•	Earth & Star Trackers	Sensor & Controller	Earth
D		2		2	3	III+	20	27.4			•	•	•	inputs		Earth
523 C		1		1	2	I-	24	12.2	•	4	•	•	•	Star Trackers	Sensor & Controller	Inertial
D		2		2	3	III	26	31.8	•	8	•	•	•	inputs		Inertial
229 C		1		1	2	Special	38	15.2	•	2				-	-	Inertial
D		2		2	3		44	35.6	•	6						Inertial
230 C		1	1	1	2	I-	46	18.5						-	-	Inertial
D		2		2	3	III	48	38.8								Inertial
16									•	1		•	•	-	-	None
DOD 1C		1		1	2	I-	26	10.5			•	•	•	Earth & Star Trackers	Sensor & Controller	Inertial
1D		1		1	3	I-	26	28.4			•	•	•	inputs		Inertial
DOD 2C		1	1	1	3	I-	64	21.8						Horizon & Star Trkrs.	Sensor inputs	Earth
2D		2	2	1	4	III+	66	37.5								Earth
DOD 3C	2			1	2	I+	22	8.6						Horizon & Star Trkrs.	Sensor inputs	Earth
3D		1		1	3	I-	24	23.6								Earth
DOD 4	1		1		2	II-	28	13.1						Horizon & Star Trkrs.	Sensor inputs	Earth
DOD 5	1		1		2	II-	30	13.4						Horizon & Star Trkrs.	Sensor inputs	Earth
DOD 6	2		2		3	II+	90	35.4						Horizon & Star Trkrs.	Sensor inputs	Earth
DOD 7A		1	1	1	5	I+	140	52.4	•	1		•	•	Horizon & Star Trkrs.	Sensor inputs	Earth
7B		4		5	C6	Special	190	671.5	•	1	•	•	•	Horizon & Star Trkrs.	Sensor inputs	Earth
DOD 8		4		5	C6	III+	190	671.5	•	1		•	•	Horizon & Star Trkrs.	Sensor inputs	Earth
					D8			225.2								

* IRCS Tankage Set = 1 Complete Set of LEM Tankage

** Refer to Figures 5.3 through 5.3-4 for power pack configurations

+Refer to additions/subtractions from these basic power pack configurations.

5.5 LEM STRUCTURAL MODIFICATIONS

Figures 5.5-1 and 5.5-2 show the extent of modifications, on ascent and descent stage structures, required to accommodate DOD and NASA flights. These structural arrangements represent the basic spacecraft structural configurations to which experiment supporting structure and bracketry is added to accommodate installation requirements of detailed experiments.

5.6 VOLUME UTILIZATION STUDIES

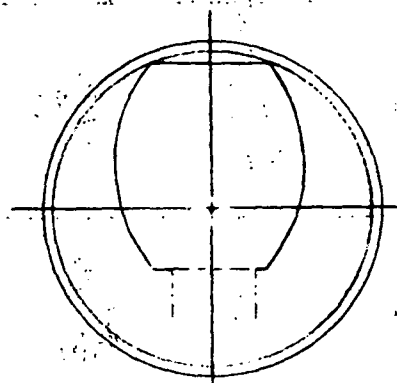
Volume available within the SLA adapter, for a LEM Lab and associated equipments, amounts to over 4100 ft³. As a general rule, there were no volume problems encountered, during this study, on any NASA or DOD experiment flights. Specific DOD flights, that were examined to identify volume-limited configurations, are discussed in Volume 3.

Table 5.6-1 illustrates the amount of pressurized volume provided for NASA and DOD flights studied. Individual Lab flights generally provide about 260 ft³ of pressurized volume. Rendezvous flights provide combined pressurized volumes varying from 357 to 750 ft³. In general, interior pressurized volume used by experiment equipment varied between 10-20%, with a couple of flights close to 28%.

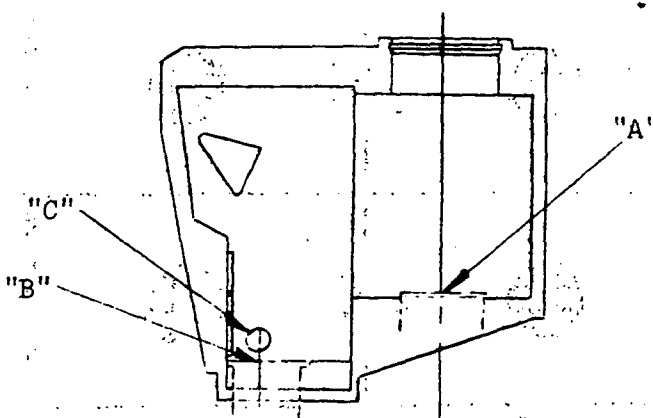
KEY:

- "A"-HOLE IN MID-SECTION FLOOR (TELESCOPE EXPER. 0702)
- "B"-HOLE IN CREW COMPARTMENT FLOOR (CORONAGRAPH EXPER. 0703)
- "C"-HOLE IN CREW COMPARTMENT SIDEWALL (VIEWFINDER)
- "D"-HOLE IN CREW COMPARTMENT FWD. BULKHEAD (VIEWFINDER)
- "E"-HOLE IN FWD. HATCH (QUARTZ WINDOW FOR SOLAR CONCENTRATOR, EXPER. 0605)

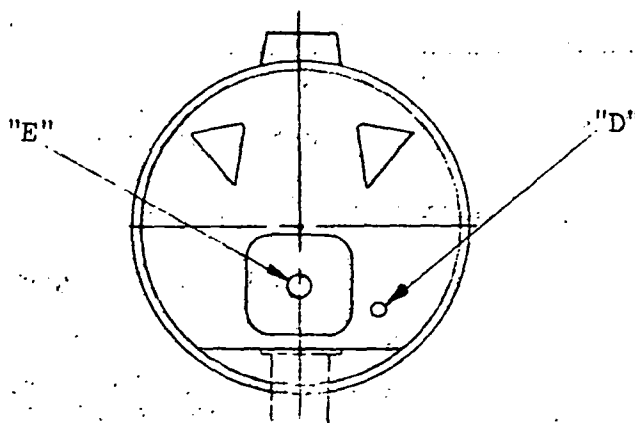
FLT. MOD	A	B	C	D	E
211-C -D				• •	
211A-C -D				• •	
507			•		
215			•		
513			•		
218-C -D					• •
516-C -D	• •	• •			
518-C -D			• •		
523-C -D		• •			• •
DOD-1C -1D	• •		• •		
DOD-2C -2D			• •		
DOD-3C -3D			• •		
DOD-7A -7B			• •		
-8			•		



VIEW LOOKING AFT



VIEW LOOKING OUTBOARD - R. H. SIDE



VIEW LOOKING FORWARD

Fig. 5.5-1 Summary of LEM Ascent Stage Modifications

View Looking Down	View Looking Up	View Looking Down	View Looking Up
 Flts 211a & 221 		 Flt 209 	
 Flts 215 & 507 		 Flt 518 	
 Flts 218 & 230 		 Flt DOD 1 	
 Flts 516 & 523 		 Flt DOD 2 	
 Flt 513 		 Flt DOD 3 	

- Key:
-  Skin Removed Top or Bottom as Indicated
 -  Side Skin Removed and Replaced with Redesigned Skin
 - Skin Removed for Access Only and Replaced

Low Profile Descent Stage on Flight Nos. 219, 211, 229, 509, 521, DOD 4, 5, 6, 7A, 7B & 10

Fig. 5.5-2 Summary of LEM Descent Stage Modifications

LEM LAB PRESSURIZED VOLUME (FT³)

Table 5.6-1

Flight No.	Basic Lab	Bio-Med Canister	Food Canister	Centrifuge	0703 Coronagraph	0702 Telescope	Connecting Tunnel	Total Pressurized Volume
NASA 209	251.5	8.5						260.0
211								
507								
509								
215								
513								
218		8.5						260.0
219				128			111	491.5
221		8.5						260.0
516					3	11		265.5
518		8.5						260.0
521		8.5						260.0
523					3	11		265.5
229							56	307.5
230	--						91.7	91.7
DOD-1	251.5		14.5		3			269.0
2			10.3					261.8
3								
4								
5								
6								
7								
8								
9								
10			10.3					261.8

Rendezvous Flights

218/219	751.5
523/229	573.0
523/230	357.2

NOTE: Basic Lab Volume = Press. LEM + ASC. Engine Cover
= 247.0 + 4.5 = 251.5 ft³